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ORBITAL ROCKETS

The following three papers were presented at a meeting of the British Interplanetary Society in London on 6th January, 1951.

I. SOME PRELIMINARY CONSIDERATIONS

By Kenneth W. Gatland*

Introduction

This Symposium can be regarded as a continuation of the survey we began with last year's lecture, "Initial Objectives in Astronautics."[1] On that occasion, it will be recalled that we opened by examining the most ambitious rocket vehicles that had received practical consideration, namely the German A–9 and A–9/A–10. Basing our arguments on the experience obtained with the A–4 rocket, we then set out to show that, using existing propellants and current engineering practice, it was possible to produce useful close-orbit Earth-Satellite Vehicles. Finally, as the main theme of that paper, we took upon ourselves a survey of the characteristics of a manned rocket vehicle capable of making a circum-lunar flight and a return to Earth. This included the application of a nuclear reactor with an assumed V_e of 10 km./sec., and it was shown that, given even this optimistic figure (remember that to-day's *Viking* rocket gives less than 3 km./sec. V_e), we could not guarantee the fulfilment of such a mission without assuming that the returning step ends in a circum-terrestrial orbit and that the crew would be retrieved by a secondary rocket which had been sent out to rendezvous with it. Our intention was not to produce a "design" for a spaceship but merely to throw into perspective the major limitations which confront us in this early formulative period of astronautics.

Fundamentally, we wanted to determine three things: (*a*) the application of nuclear-energy as a primary propulsion unit; (*b*) the use of chemical propellant in a final step which serves also as an economical shielding pattern during the operation of the nuclear reactor, and (*c*) the influence of expendable construction. Our recommendations were to continue the theoretical studies of expendable-tank construction, as a method of improving the structural-factor, and to develop the orbital rocket as a first step towards interplanetary flight.

We do not wish to take up your time to-night with a lengthy technical discussion of the Earth-Satellite Vehicle, but merely intend to sketch in the background of this vast subject, which, in company with other delegates, we hope to consider at greater length at the Second International Congress

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of Astronautics in September. Here, we shall be examining the orbital rocket in the form it is likely to appear during the initial space-flight ventures, before the advent of the elaborately equipped "space-stations" proposed by von Pirquet, "Nordung," Oberth and Ross, which can arise only when space-flight is fully established.

The Course of Development

When our Society was reformed in 1945, I think none of us would have considered that military research would provide within a few years the first real springboard to the planets.[2] We recognized the value of the A–4 rocket and the vast fund of experience and resources that had been built up in Germany but there was still no real indication of how the rocket-weapon would evolve towards the spaceship. It was left to the Americans to exploit the A–4 for high-altitude research and then to go on to produce their own rockets expressly for this purpose. Meanwhile, the British authorities at Cuxhaven, having reconstructed and fired some A–4's at targets in the sea, concluded that the long-range rocket was inaccurate and consequently of small value. It has been ignored by them ever since, but sueh has become the urgency in recent years for precise knowledge of upper atmospheric conditions to supplement our own research studies (particularly in nuclear physics), that investigators are slowly coming to acknowledge the rocket's value to science. To-day it would be no more surprising to read in the daily papers that *Aerobee* rockets may shortly be imported into this country for basic research purposes than that American *Sabre* jet-fighters are to enter service with the R.A.F.

Of all the rocket vehicles developed for the American high-altitude programmes, the *Aerobee*[3] has undoubtedly proved the most useful and it is now being produced in large numbers. By means of the U.S.S. *Norton Sound* (a sea-plane tender converted for rocket-launching), it has been fired elsewhere than the White Sands test-area—from various locations in the Pacific Ocean. In a paper[4] presented in March, 1950, at a meeting of the American Rocket Society in New York, Dr. J. A. van Allen emphasized the need for obtaining data from a large number of widely dispersed areas. It is vital, he said, that reliable and inexpensive rockets are developed to enable research groups using different techniques to expand and verify upper atmosphere data collected from sites in other parts of the World. Only then can really accurate knowledge of the ionosphere and the extent and nature of radiation phenomena be obtained. This is, perhaps, an indication of how research may be extended in the not-too-distant future.

Whatever the reasons for the apathy on the part of officialdom here to exploit the results of German research in the shape of high-altitude missiles (which, at the same time, would have contributed a fund of data on all manner of problems related to ground-to-air rocket weapons), the matter cannot be dismissed on the grounds of economy alone. With our universities brought into the picture, a British high-altitude programme could have begun as a branch of existing military research before the guidance techniques required by the A-A weapons were perfected. That primary missile would not only be of value to us scientifically, but, with it, the military fraternity could have carried

out all their basic research—on stability, inherent control, step-techniques, etc. —at an early stage, besides obtaining experience in the handling and servicing of practical rocket vehicles. From that elementary basis could have been developed the fully-fledged guided-weapon. If such a policy had been adopted in 1945, when the British Guided-Missile Programme was laid down, I very much doubt whether this would either have impaired our economy or the pace of our research effort; indeed, as regards the latter, such a procedure might easily have accelerated it.

I have stressed this point because many people appear to wonder why, when the Americans are doing so much to sponsor high-altitude research within their own military organizations—and, furthermore regard it as vitally important—the same procedure is not being followed here. At the present time, having lost the initial advantage of incorporation in the military programme, it is now undoubtedly a matter of economy, but necessity should ensure that "scientific rocketry" is not entirely lost to Britain. We cannot, unfortunately, hope to compete with the United States to the extent of developing the Earth Satellite Vehicle, but in later years, as the scope of this enterprise broadens, it may possibly lead to collaborative research between the two nations. We may hope so!

Orbital Techniques

The significance of the orbital rocket in the evolution of astronautics has been made increasingly evident since the war by paper investigations, which have shown that the most potent chemical propellants are inadequate for interplanetary missions beyond the stage of the pilotless missile. Until recent times, the initial ventures in interplanetary flight had always been considered in terms of point-to-point missions, such as we find in the film, *Destination Moon*.

In October, 1948, two papers were submitted independently to the Society, which came to broadly similar conclusions: one was "Orbital Bases"[5] by Mr. H. E. Ross, and the other, "Rockets in Circular Orbits"[6] which represented some views of my own. Both set out to show the advantages to be obtained in breaking down a given space-flight mission into well-defined stages and introduced the expression, *Orbital Technique*. In its simplest form, this technique amounts to setting up a rocket in a circular orbit around the Earth and refuelling it at that location from secondary "tanker" rockets. Why this is proposed will be obvious after a moment's reflection; it is based on the elementary principle that it is easier for a bricklayer to carry a load of bricks a specified distance by making two or more journeys, than for him to attempt to carry them all in one. Similarly, in terms of space-flight, it is far easier to attain our desired "escape velocity" if we build up to it in several stages.

The idea then is simply, that pilotless rockets build up a store of propellant in a stable orbit around the Earth, this representing a substantial energy-potential towards attaining an interplanetary mission. If these tanker rockets were located 500 miles from the Earth's surface, the propellant would be brought into the sub-orbit and stored at a velocity of 7·5 km./sec. (approximately 18,600 m.p.h.).

It would be rash, however, to suggest that Orbital Technique is the panacea for all our space-flight troubles. One is brought abruptly back to reality when one reflects that, to place a mere 350 lb. of useful payload in a close-orbit with available propellants, will require a vehicle of 76 tons initial mass, and, as we shall see in the final paper, this assumes quite favourable design parameters. When we consider bringing not *pounds* of *instruments* but *tons* of *propellant* into the sub-orbit, the problems which still confront space-flight are thrown into bold relief.

At an early period of our investigation, we considered the case of an orbital refuelling vehicle assuming a chemical propellant with a V_e of 4 km./sec. (liquid oxygen/hydrogen). With two-steps and a system of expendable-tanks, the all-up weight came to 1,000 tons, the cost of bringing 10 tons of propellant to orbital velocity. Comparative figures for an atomic-powered vehicle (similar in conception to the Circum-Lunar Rocket of our previous paper,[1] but without the final chemical step-rocket), with a payload of 60 tons, gave an initial weight of 1,200 tons. This involved the use of a two-step chemical booster, but a further conception of an entirely atomic-rocket (10 km./sec. V_e—direct ascent without booster), brought the all-up weight down to approximately 550 tons for a 50 ton payload.

With these figures in mind, we see that, far from making the project *more* economical (as some people appear to imagine), the initial interplanetary venture would be made with greater overall effort, greater technical complexity, involving the expenditure of much more propellant, than if the project could be sponsored as a "one-shot" effort. Space-flight will be a costly game however it is played, but, since, in any case, a direct flight to our nearest planetary neighbour, cannot be made even assuming the existence of an atomic-reactor capable of 10 km./sec. V_e, orbital techniques are a realistic means of spreading the "energy load" in such a way that, if we are prepared to take the trouble, we could achieve space-flight with a V_e of half that figure.

There are many people who have said that this additional complexity is out of all proportion to the objective—the project's "real-estate value" as the Americans would call it!—but there is just one vital point that should not be overlooked. In developing this elaborate "space-shuttle," we shall be building up a practice that will pay increasing dividends with each new venture across space and, in the long-term view, this will be much more beneficial than if we begin with one mighty rocket that just manages to crawl home.

Mr. Ross has already dealt with the possibilities of an Earth-Moon-Earth flight, assuming a chemical propellant with a V_e of 5 km./sec.,[5] which, without Orbital Technique, is clearly impossible. Although the novel ideas he introduced could easily reach fulfilment in Man's initial venture across space, it is our belief that interplanetary fight will not really come into its own until we have achieved *both* an applied form of atomic-energy and a system of orbital refuelling.

These ideas and, indeed, the whole conception of space-flight—even when the elements of the subject are explained—seem very much a closed-book to many engineers who should know better. A short while ago, I had an opportunity to judge this for myself in a conversation between two of my colleagues.

"Did you see Arthur Clarke's latest effort on television?" said the first. "Yes, I thoroughly enjoy those programmes. A bit fantastic though, that odd-shaped space-ship he used this time—the one that's supposed to hang out in space and be refuelled!" It was the reply that was so disarming! "Yes, some of those on the covers of '*Startling Stories*' seemed much more practical!" I naturally went into the breach on Mr. Clarke's behalf and though I cannot claim that my friends went away as fervent supporters of Orbital Technique after the short discussion we had, they received (I hope!) a better appreciation of our reasons for departing from the classic style of spaceship to one which looked peculiarly like a dumb-bell.

Although Oberth undoubtedly laid the theoretical basis for the spaceship, credit for the first engineering analysis is due to the pre-war Technical Committee of this Society. The Cellular Rocket,[7] as the conception was called, focussed attention upon the major design problems that would have to be faced and suggested many solutions. But, perhaps most of all, it corrected the widespread belief that spaceships must be magnificent streamlined creations with fins and rows of portholes. If ever the covers of science-fiction do spring to life, it will not be until one of their mythical "hyper-drives" has become a reality and we are freed forever from our pressing mass-ratio problems; until that great day arrives, we must, I think, continue to rely on our high-density spaceship with its modest payload.

The Composite Rocket

I do not propose in this introductory paper, to deal with the initial minimum payload rockets—you will hear our conclusions about these in the following papers—but since the whole conception of orbital rockets seems so much of a mystery, let us first of all examine the advantages one may logically expect from their future development—in particular, since the planets are our objective, the man-carrying variety.

The type of rocket we envisage is a composite vehicle comprising a main propulsion stage, incorporating the atomic-reactor, and a small, secondary, "chemical rocket" which contains the crew. These separate vehicles are first established in a sub-orbit of the Earth, being driven out as complete units, or, possibly, in the case of the atomic section, assembled in space from prefabricated sections. The "atomic stage" of the composite rocket would be employed purely as a means of propulsion between planets and enters a sub-orbit at both ends of the flight while the minor rocket, which contains chemical propellant and the landing equipment, leaves the parent vehicle and descends to the surface.

After the mission is completed, the minor rocket returns to the sub-orbit and re-links with the parent vehicle, which propels it back to a circum-terrestrial close-orbit.

This conception seems satisfactory until one recalls that different landing techniques may be involved in operations between characteristically different planets. In the case of a Moon-flight (because of the absence of any real atmosphere), this means a vertical descent using reverse rocket braking in conjunction with a radar-altimeter and landing-legs, whilst the final return to

Earth will probably require both rocket and aerodynamic braking methods of approach which involve quite different design conceptions in what we should ideally regard as the same vehicle. It is conceivable, therefore, that a form of supersonic rocket-glider—similar to the German A–9—previously brought up into the sub-orbit, would be employed at the Earth end; the crew would be transferred into it after the two vehicles had been brought to proximity. However, this is not the hardship it appears at first sight because, with a third vehicle returning the crew to Earth, not only does this introduce a useful safety factor (the winged-vehicle being designed for the *express* purpose of navigating an atmosphere), but there is the added not inconsiderable advantage that the *complete* composite rocket is retained in space ready for further use. Since, in any case, it will be necessary to perfect the art of transferring propellant and crew-members from one rocket to another, this does not introduce any further complication.

When anyone suggests refuelling rockets in space and moving members of the crew from one rocket to another, too often this appears to stretch the imagination of the layman to the elastic limit. Most of the scepticism, of course, comes from ignorance of the principle of satellites and many people have the idea that one rocket will be engaging in a long and hazardous chase of another in order to make the vital connection, *Meteor-Lincoln* fashion. In effect, once orbital-flight is established, the actual link-up of the two space-rockets will be achieved, certainly with no greater difficulty than the smooth operation already demonstrated by Flight-Refuelling Ltd., in the atmosphere.

What one has to bear in mind, first of all, is that there is no "chase" in the accepted sense and absolutely no question of buffeting—the objective will be to place the out-going rocket in a matching orbit with the vehicle already circling the Earth.

Now is not the time to go into the mathematics of the question—we hope to do that in September—but briefly, the most economical approach trajectory will be as follows: take-off vertical, gradually easing from vertical into the curve of synergy which brings the vehicle into an introductory or preliminary orbit at a certain optimum altitude below the desired orbital height. At that altitude, the vehicle would maintain itself temporarily in the state of free gravitation without further expenditure of energy. This period would serve for adjustment of the orbit to ensure, in the case of a rendezvous between two orbital vehicles (the first already in the orbital position) proper co-ordination with the "satellite," which the ascending vehicle is approaching because of its shorter period of revolution. Thrust is then applied for a short interval to accelerate the vehicle from the lower orbital velocity to the velocity corresponding to the perigee of the transfer orbit (i.e. the ellipse whose lowest point is situated on the introductory orbit and whose peak—apogee—is situated in the final, "upper," orbit. The vehicle completes approximately half a revolution of the Earth and during this period fine adjustment will be given to its velocity so that it will arrive in the required direction at the correct orbital height. To complete the operation, it will then be necessary to accelerate the vehicle from the velocity possessed at the time of reaching the peak of the transitory ellipse to the final orbital velocity.

The successful application of this technique naturally presupposes the development of quite intricate guidance and control equipment but even this does not show up too badly when the problem is considered against its true background. It must be remembered that orbital or satellite rockets will exist in a condition of free-gravitation and that so long as the location system is sufficiently accurate for two vehicles to be placed into a matching orbit, and homed to proximity, their relative speeds can be brought to zero. An observer looking out from one vehicle to the other would have no apparent sensation that the two were in motion at all, and only the rotation of the Earth and, of course, the slowly changing aspect of the heavens would subsequently be noticed. Actual linkage might then be achieved with no more elaborate equipment than a small line-carrying rocket shot from one vehicle to the other, perhaps homing automatically into some form of "target cage." That achieved, not only could propellant transfer be made and space-suited crew members exchanged, but two components of a composite rocket might fairly easily be joined. If necessary (assuming zero "g" has no disabling effect), it would be possible for the crew to work outside the vehicles and to move themselves around the area with the aid of cold-type reactor units. In this condition where material is "weightless," considerable feats of engineering assembly could be achieved by one or two men working for quite short periods. It may, for example, be a more economical proposition to transfer *tanks* of propellant than to employ pumps and pipe-lines, and the expendable-tank rocket fits ideally into this conception since the tanks are designed as load-carrying structures and could easily be made interchangeable among the various types of rocket. Certainly this method would be the more efficient because in any pipe-line system there are bound to be losses, and, of course, we are conserving our pump-fuel.

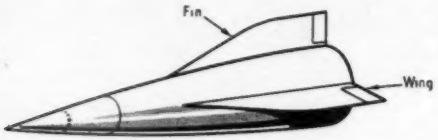
The unique properties of the sub-orbit are not widely appreciated and although their consideration, at first sight, may appear akin to "science-fiction," they are nevertheless based on well-established principles.

The Future, and Orbital Refuelling

Finally, let us consider this prospect of orbital re-fuelling in the light of an illustration. I should here like to express my thanks to the publishers, Weidenfeld and Nicolson, Ltd., for permission to use two drawings from the book *Rockets Into Space*, which I am preparing for publication later this year. This book contains nine chapters, each by a well-known authority in the rocket and astronautical fields, and the following conception is due to our own Mr. A. V. Cleaver.

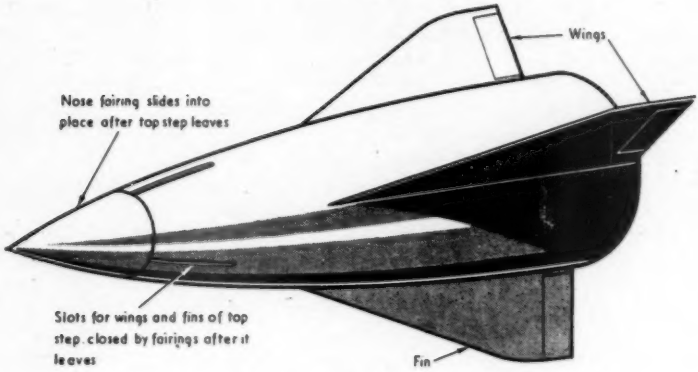
Propellant and payload, as we have seen, would be brought out from Earth in "tanker" rockets, which will rendezvous in a circular orbit beyond the atmosphere with the main vehicle, and there refuel it. The "tanker" rockets can conveniently be referred to as Type "A" (Fig. 1) and the main vehicle as Type "B" (Fig. 2).

The Type "B" will start its motors after refuelling and travel to, say, the planet Mars, finally decelerating to take-up a stable orbit around Mars. It will then transfer its payload to a rocket of a third type, Type "C" (Fig. 2); this will make the actual landing.



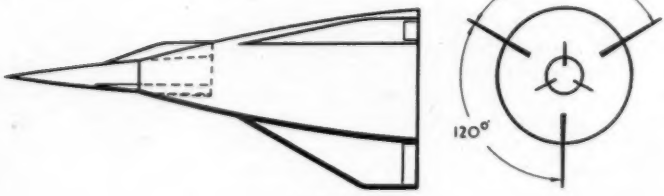
TOP STEP

Take off would be vertical, so that under these conditions, wings and fins of all steps would merely act as fins, in line of flight


Wings
Nose fairing slides into place after top step leaves
Slots for wings and fins of top step closed by fairings after it leaves
Fin

LOWER STEP

Actually there will probably be several lower steps.



ASSEMBLY

TYPE 'A' SHIP

Drawn by J. W. Wood

FIG. 1

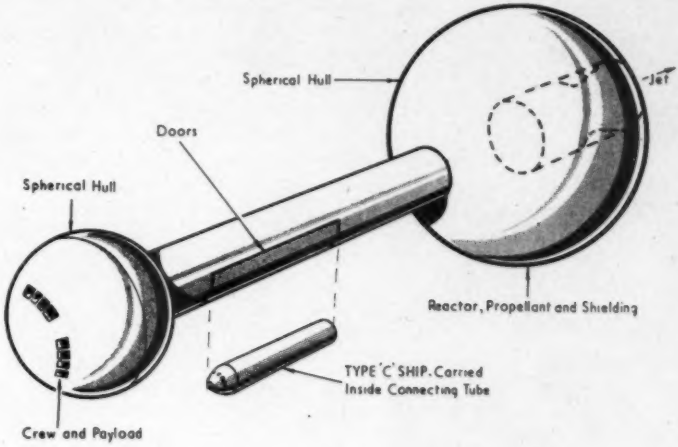

Spherical Hull
Jet
Doors
Spherical Hull
Reactor, Propellant and Shielding
TYPE 'C' SHIP. Carried Inside Connecting Tube
Crew and Payload
TYPE 'B' SHIP
Hemispherical Nose
Cylindrical Hull
TYPE 'C' SHIP
For case of landing on Moon, or on a small planet with negligible atmosphere. No attempt at streamlining, no wings or fins, perhaps only one step necessary
Drawn by J. W. Wood

FIG. 2

Subsequently, for the return flight, the Type "C" will take-off from Mars and rendezvous with the Type "B" once more in its "stand-off" orbit around that planet. The Types "B" and "C" will then return to the Earth orbit, where they will be met by the Type "A," capable of landing the payload back on Earth.

Such a scheme would enable the various rocket vehicles to be specifically designed for their own particular duties. These are so different that great diversities in the appearance and mechanism would emerge. For example, the

Type "A" would probably retain chemically-powered motors to avoid radioactive contamination of the launching site and lower atmosphere. It would also probably employ wings to facilitate landing back on Earth as a glider, and it would almost certainly be of multi-step design.

The Type "B" would be radically different. It would never land on the surface of any planet and indeed would probably have been actually constructed out in its terminal orbit from components taken out there by freighter rockets. Operating always in free space, it would employ nuclear power. It might also use the technique of accelerating to the velocity required for its final voyage-orbit at a low rate, by the continuous application of a relatively low thrust for a fairly long period (as discussed by Messrs. Shepherd and Cleaver in their paper, "The Atomic Rocket,"[8] and by Oberth in "The Electric Space-Ship."[9])

Certain other interesting possibilities present themselves. If a low acceleration flight plan were adopted, the structure of a Type "B" vehicle would never be subjected to high stresses and could therefore be of a character appearing extremely light by any terrestrial standards, even those of aircraft construction. Moreover, since it would never be required to penetrate any atmosphere, no considerations of streamlining whatever need influence its shape. Clarke has suggested it might take the form of a "dumb-bell"—two spheres joined by a connecting tube.[10] The spheres, of course, represent the ideal structural shape for a pressurized vessel; one could house the crew and payload, the other the nuclear power-plant and propellant. Separation of the two by the connecting tube would be desirable from considerations of shielding the crew from the harmful radiations of the motor in the other sphere. Inside the tube, the Type "C" landing rocket, with the necessary airlocks for boarding it, might be carried.

The Type "C" would probably be a chemical rocket similar to Type "A" but of size, number of steps, mass-ratio, etc., differing from it and determined by the gravitational field strength of the destination planet. Whether it was winged, or not, of course, would depend on the presence, or otherwise, on that planet, of an atmosphere having a sufficient density to justify the use of aerodynamic braking.

Now that we have surveyed the general picture and obtained some appreciation of the overall problem, it is opportune to investigate the more immediate applications for orbital rockets.

In our final paper, Mr. Kunesch will be giving you an impression of a close-orbit Earth Satellite vehicle with modest payload. But, since the structural factor is all important to the performance of any rocket, it is essential that detailed consideration be given to the various methods of construction and their respective weight penalties, which is our reason for including a structural analysis in this Symposium.

REFERENCES

(1) K. W. Gatland, A. E. Dixon and A. M. Kunesch, "Initial Objectives in Astronautics, *J.B.I.S.*, **9**, 155–178, July, 1950.

(2) J. V. Forrestal, *First Annual Report to U.S. Congress on the National Military Establishment*, December, 1948.

(3) J. A. van Allen, L. W. Fraser and J. F. R. Floyd, "The Aerobee Sounding Rocket—a New Vehicle for Research in the Upper Atmosphere," *Science*, 31st December, 1948, **108**, 746–747.

(4) J. A. van Allen, "The Use of Rockets in Upper Atmosphere Research," paper read at a meeting of the A.R.S., New York Section, 17th March, 1950.
(5) H. E. Ross, "Orbital Bases," *J.B.I.S.*, **8**, 1–19, January, 1949.
(6) K. W. Gatland, "Rockets in Circular Orbits," *J.B.I.S.*, **8**, 52–59, March, 1949.
(7) R. A. Smith, "Lunar Space-Vessel," *Flight*, 12th February, 1942, 134 (*d*), (*e*) and (*f*).
(8) L. R. Shepherd and A. V. Cleaver, "The Atomic Rocket," *J.B.I.S.*, **7**, 185–194, September, 1948; 234–241, November, 1948; **8**, 23–37, January, 1949; 59–70, March, 1949.
(9) H. Oberth, "The Electric Space-Ship," *Radio-Electronics*, Part 1, December, 1950, 32–4; Part 2, January, 1951, 74–82.
(10) A. C. Clarke, *Interplanetary Flight*, Temple Press, London, 1950.

II. THE ROCKET STRUCTURE, WITH SPECIAL REFERENCE TO EXPENDABLE CONSTRUCTION

By Alan E. Dixon*

Introduction

To-day one can go into a shop and buy a book on "Aircraft Structures" but it is most unlikely that books on rocket structures will be available for many years to come since, as a specific engineering subject, it is as yet unborn. The structures of rockets in use to-day largely follow the practice of modern aircraft design, giving rise to similar problems and to a degree, similar solutions.

In recent years, however, we have seen the development of new techniques in the theory of rocket design which have brought with them new structural layouts and conceptions and, in turn, a host of new problems which require solution if these new techniques are to bear fruit in practice. Through the medium of these problems and their solutions, the science of rocket structures will find expression as an independent subject.

A most important new structural technique is due to Mr. K. W. Gatland and is known as Expendable-Tank Construction. This will already be familiar to those who have read "Expendable Rockets"[1] and our recent paper "Initial Objectives in Astronautics"[2] and I do not propose to refer at any great length to the *advantages* to be gained from this method of rocket construction: these, I think you will agree, can be regarded as conclusive!

It is rather my purpose to show how this original idea has been developed during the past two years of collaborative study, to examine some of the problems and illustrate certain possible solutions. This is particularly so in regard to the effect of expendable-construction on the performance of the Orbital Rocket, the subject of our concluding paper.

The Load System on a Rocket

The load system acting on a rocket is primarily determined by the thrust applied by the motors. In the first place, the thrust decides the maximum end loads in the structure; the motors, their type and mode of operation determine the propellant feed pressures and hence, to a point, the tank pressures.

The additional bending and shear loads on the structure are fixed by the designer in deciding on the amount of latitude to be given to the rocket's controllability and to the sensitivity of its control system.

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To enable us to visualize these matters more clearly we shall consider a rocket during the initial moments of take off. As the fuel is fed to the motors the pressure in the feed system is built up and the pressure load in the tanks is established.

Simultaneously, the motors proceed towards their full thrust values and the rocket in turn begins to accelerate.

During the vertical ascent, the rocket will have to perform steering manoeuvres to correct possible deviation from the flight path or to take the rocket out of the vertical path into the orbital trajectory. Inevitably, manoeuvre loads due to the steering of the rocket will be applied. In order to change the direction of flight it is necessary to rotate the rocket until its line of thrust lies in the new direction. This is done by applying a side thrust at the base of the rocket. This rotation produces lateral inertia loads which in turn develop the shear and bending moments on the rocket structure.

The side load is usually supplied by the deflection of guide-vanes mounted in the jet efflux of the motors. The actual value of the shear and bending moments produced depend on the rate of rotation, and the distribution of weight in the rocket. This rate of rotation in turn depends upon the magnitude of the side thrust, its distance from the centre of gravity, and the moment of inertia of the rocket.

In the case of a large rocket an additional factor influences the loads applied to the structure. This is the slight but inevitable flexibility of the structure. Some idea of the effect of too much flexibility may be gained from the following considerations which have been exaggerated a little to make the case clear.

If we consider the imaginary centre-line of a rocket moving along some predetermined course and that a side load is applied such that rotation is caused; then, due to the inertia effects, the structure will bend and the centre line will now be curved. In this condition the thrust will act along a line which is tangential to the curved centre line with the result that the thrust is offset and produces its own moment about the rocket C.G. somewhere along the centre-line. The point is that which ever way the additional moment acts, the net moment on the rocket will not be the value actually calculated.

Weight Distribution

Before proceeding with a discussion of the rocket structure it will be advisable to consider the weight distribution, since some of our arguments must hinge on this assumed distribution.

The gross weight of a rocket is contributed by the payload, the propellant, the propellant tanks and insulating material, the rocket motors (with their pumps and turbines) and the ancillary equipment to provide the various services. The propellant weight and, to a large degree, the motor weight are fixed and cannot be varied. Structural techniques and pump systems, on the other hand, offer the designer considerable scope for ingenuity and promise to effect a substantial improvement in the overall performance.

This is especially true with expendable construction in which *step-technique*, the *rocket structure*, and the *tanks* have been combined in *one inherent conception*.

Rocket Layouts

The normal types of rocket of to-day, which fall mainly into what I shall call the monolithic class (since these rockets have the same structure weight throughout the duration of flight), have layouts which are largely identical. Generally speaking they follow the pattern laid down by the V2 design.

The streamlined nose contains the payload of instruments. In the centre section are contained the propellant tanks and usually the "brain" of the rocket, the rear section being devoted to the propellant pumps and the motors themselves.

The centre portion usually contains two or more large tanks of suitably light construction which are rigidly attached to the main structure. The tanks are designed to take the pressure loads only, all end and bending loads being carried by the main outer structure. With fluids such as liquid oxygen, and more particularly liquid hydrogen (the boiling points of which are in the order of −300° F. and −400° F. respectively) the problems of heat transfer and insulation are not exactly small, and in the case of the V2 the space between the tanks and the outer shell is packed with glass wool.

The structure of this type of rocket presents few real problems unless of course the rocket is very large and heavy, for as the size goes up so certain minor problems tend to become major ones; and whilst they could be ignored in the small rocket, they can assume great importance in a larger design.

As an example we may take the longeron members running the whole length of the rocket, from the payload down to the motor thrust ring. These act as struts and their load carrying capacity depends on the moment of inertia of their cross section and on the square of their lengths. This means that if the rocket was twice as long, without additional support their load carrying capacity would be $\frac{1}{4}$ the original load. To overcome this either the longerons would have to be made much larger in cross section or their effective lengths reduced by fitting frames or bulkheads inside. These are often placed between fuel tanks, which brings us to a further example.

As the size of the rocket goes up, often with its consequential increase in diameter, these frames become more flexible and whereas in a small rocket they can be designed from general engineering experience, this can no longer apply. In the larger vehicle, flexibility would have to be investigated to ensure that the frames had sufficient stiffness to supply the fixing load to the longeron members and enable them to do their job.

The expendable-tank rocket gives much greater scope in variation of layout than the normal type and it is with this conception that I now wish to deal. The layout, in turn, determines the shape of the tanks and, as we shall see later, this shape, together with the general configuration, determines the tank weights and also the particular bias of the tank towards various problems and design techniques.

The expendable-tank rocket, by virtue of its conception, requires that the propellant be stored in a single tank unit. The ideal, of course, would be to use a mono-propellant. With a bi-fuel rocket, however, these tanks must be partitioned, and from considerations of symmetry an annular partition would

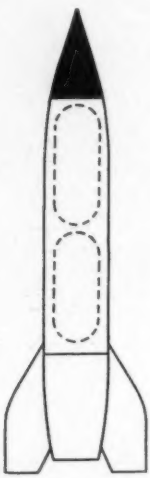


FIG. 1.

FIG. 2.

FIG. 3.

Diagram of payload disposition in various rocket configurations: Fig. 1, Orthodox—typified by German A-4; Fig. 2, Simple expendable-tank construction; and Fig. 3, Step construction with annular expendable tanks in 1st step.

be the most suitable. We can now refer to Fig. 2. This layout is derived from that of a conventional bi-fuel rocket, the main difference being that the payload has been moved from the nose to a position immediately in front of the motors, which leaves the complete forward section free for propellant in expendable-tanks. Incidentally, it is possible that the first expendable-tank rocket may be of this particular layout since it is relatively simple to design and build. Fig. 4 shows the tank which we refer to as the "split" type, the two halves being held together by tension bolts. The oxidant is contained in the inner tank whilst the fuel is contained in the annular outer section. The tank ends and additional structure are omitted for reasons of clarity. In Fig. 3 we have a typical layout of the type generally envisaged to-day, which is somewhat more complex in design and would be suitable for fairly large rockets. From the diagram it can be seen that it consists of an internal rocket structure of the V2 type, with the expendable tanks wrapped round it to form a second hull or shell. In this case the motors may be an integral part of the inner structure or they may be attached in such a way that they can be jettisoned, the inner rocket continuing after separation under its own power.

Inspection of the layout shows that the tanks are annular shaped and it is evident that the design can become quite complex when they are constructed to contain both fuel and oxidant. Note also the cone shaped ends to the tanks. This type of tank presents special design problems which will be discussed later.

Various Fuel Supply and Pressurising Systems

There are various methods of getting the fuel from the tanks to the motors and in some rockets (in fact, in most rockets), a combination of more than one method is used. The basic problem is to boost up the propellant pressure sufficiently for injection into the combustion chamber.

One method is by pressurizing the tanks, there being a straight-through pipe to the motor or motors, the pipe itself being fitted with a control valve. A second method is to suck the fuel from the tanks and build up the pressure by means of pumps. Such a system would require an adequate ventilation system for the tanks and for very low temperature propellants this might prove difficult.

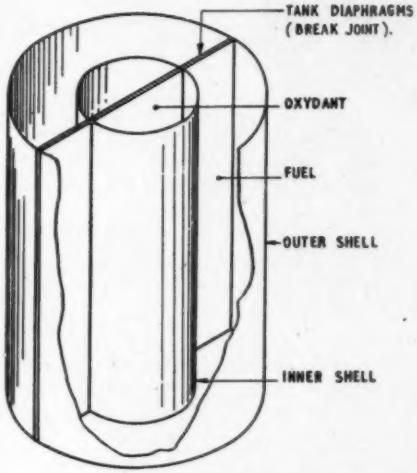


FIG. 4. Schematic diagram of simple expendable-tank construction (see rocket conception, Fig. 2).

A more usual method is a combination of both these systems, the tanks being lightly pressurized and pumps utilized for the remainder of the work. The pumps will probably be driven by small steam-turbines, but larger pumps would require turbines of the same order as the turbine in an aircraft jet-engine. With larger rockets, calculations indicate the need for pumps using enormous powers, these in turn requiring equally large turbines. Under these circumstances it is possible for development to be retarded not alone by the question of motor design, but by the problem of pumps and the necessary turbines. In fact, the whole project might hinge on this score alone, especially with the use of very low density fuels such as hydrogen.

It might be easier, therefore, to produce large volume "idling pumps" whose job would be to supply the last 50 lb./in.^{2} or so to enable the fuel to be injected

into the motor. The balance of the pressure would be supplied by pressurizing the tanks. It might be argued that this would greatly increase the weight of the tanks and although we are prepared to concede this there is, at the same time, evidence that there may be considerable advantages in this method.

It must be realized that a large pump together with its turbine and accessories can be a very weighty item and will be carried during the entire flight. If idling pumps were used, although they would still be fairly massive, the turbine requirements would be much less, affecting a reduction of the weight and complexity of the installation. A greater proportion of the weight could then be concentrated in the tanks themselves which are periodically jettisoned instead of contributing merely "dead-weight" over the entire period of the flight. Later we intend to show that these heavy tanks will be less troublesome than the lightly constructed variety.

A further advantage of the high-pressure, thick shell, tank, is that damage during handling is less likely to occur than with the lightly constructed low pressure kind.

Types of Tanks and Design Problems

Let us now examine the design problems of individual tank shapes, proceeding from the annular type already referred to in connection with the expendable layout shown in Fig. 3. However, before so doing, it is necessary to underline the essential differences between the structure of the expendable-tank and monolithic rockets. In the case of the monolithic rocket, a separate outer structure is provided to take the bending and end loads. With the expendable-tank rocket, there is a single structure which must be capable of carrying the bending loads as well as containing the propellant.

Since the outer tank skins are required to carry the end loads, joints must be provided between the individual tank units and since these joints must be of the quick release type it is obvious that we cannot have too many of them. Because of the mode of operation, these tanks must have fuel transfer valves at each end and the conical ends of the tanks enable the valves to dis-engage when the tanks are thrown off. Before discussing the transfer of loads from one tank unit to another we must consider the loads acting on the tanks themselves and the way in which these loads are carried. We shall first consider the annular-shaped tank unit.

The outer wall or skin of such a tank presents no problem from the point of view of pressure loads. The inner wall, however, is a different proposition. The resistance offered by a thin walled tube to crushing by externally applied pressure is virtually negligible. Means must be found to enable this wall to withstand the applied pressure loads and such a method was described in our paper last year[2]; briefly the idea is to provide a series of radial baffles inside the tanks joining the inner and outer walls so that any attempt by the inner wall to collapse is resisted by tension in the baffles, these loads being transferred to the outer wall.

The long narrow curved panels between the baffles are further supported by hoop members round the inner wall and at right angles to the baffles. The main purpose of these hoop members is to reduce distortion of the tank since

the pressure acting on such a large unsupported area of plate would cause excessive dishing of the skin between the baffles. This problem does not exist in the simple design shown in Fig. 2 since the radial partition has design conditions which do not exist for the inner wall of an annular shaped tank. There are, of course, other ways of supporting this inner wall.

Another method would be to use closely spaced stiffeners running fore and aft between the tank ends. These stiffeners would in turn require heavy circular frame members at each end of the tank and it appears that the inner wall, whatever its methods of construction, will exact a large weight penalty. Calculations on several hypothetical tank units of the annular-type indicate that the weight of the inner wall will vary from three to ten times the weight of the outer wall under the same pressure conditions. The actual ratio of these weights will naturally depend to a certain extent on the ratio of their diameters.

So far we have only considered the tank walls under fuel pressure loading but with these tank units the outer shell must also carry all end loads due to bending, etc. Unsupported shells are prone to buckling under compression and hence it is necessary to support the skin by longeron members. The size and shape of these longerons depends upon the individual design, the number of longerons, and the magnitude of the loads involved. There are also certain relationships between the geometry of the skin or shell to be supported, and certain properties of the longeron which can be determined without the magnitude of the loads being known.

One such relationship is the ratio of the moment of inertia of the longeron section about its neutral axis to its moment of inertia about the median line of the skin it is supporting. The other relationship is the ratio of the longeron area to the area of the skin being supported. This is also bound up with another relationship connecting the edge stress and average stress conditions of the skin. When considering the shape and size of the longerons, it must be borne in mind that they are jointed at the ends of each tank unit and that the load line of the joint must coincide as nearly as possible with the load line of the longeron to reduce any local bending that might occur. The section envisaged for these longerons is the top hat or channel section.

From these considerations the geometry of the longeron can be determined, and the area of the longeron will have to be something of the order of twice the area of the skin it is supporting. Thus, from the weight distribution point of view, it will be reasonable to state that the weight of the longerons will be twice the weight of the outer shell. Before leaving the longerons, we shall deal briefly with the longeron joints. These joints must be capable of transferring the loads from one tank unit to another. They must also be of the quick-release type.

The joint envisaged would be multi-pronged. We envisage that the twin pins of the joint might be withdrawn by the action of high-pressure gas generated by a cartridge explosion, the action being similar to that of a Koffman aero-engine starter. It should be noted that although these joints will operate under load, in all cases the loads will be due to the empty tank about to be jettisoned. This is perhaps one part of an expendable rocket structure which will require mainly experimental and test work rather than extensive theoretical investigation.

We might also include in this category the explosive tension bolts which hold the two halves of each tank unit together. Although their primary work is to bind the tanks together, as tension fittings they have an additional role to play. The end-loads in the longeron members due to bending, which results from manoeuvre loads on the rocket, are the direct result of shear in the rocket structure skin. The distribution of this shear in the skin is such that at one point the shear has zero value and at another point a maximum value. Under certain conditions it is possible for this point of maximum shear to coincide with the tension joint between the two tank halves, in which case, these tension bolts will also be in shear, thus complicating the internal loading of the bolts and hence the design.

Problems on Construction

The construction and assembly of an expendable-tank rocket presents many problems some of which, as we have seen, become more acute as the size of the rocket increases. But it should be remembered that this applies to the more orthodox designs as well. Before going any further we shall consider some of the requirements to be satisfied by the tank units.

In the first place they may be considered as high pressure vessels; and secondly, they will, in general, contain a low temperature liquid. My personal opinion is that these two conditions, when combined, rule out the use of riveted joints.

Probably all joints will be welded. Where additional structural members are required to be attached to the skin, several alternatives are available. These members or fittings might possibly be bonded to the skin with some metal-to-metal adhesive. If this is not possible, and the skin is fairly thick, then some type of "blind" riveting might be employed. Here the rivet hole is only drilled part of the way through the second member. A special tool is used whereby an internal thread is produced; the rivet is then inserted and the shank expanded into the threaded hole.

If the additional structural members on the shell are in the form of shallow ribs, it would be possible to machine the combined skin-rib combination from one thick block. If it is required to have members running across a sheet of metal in two directions and intersecting, it might even be possible to cast the complete unit. This method could be applied in a case where a curved sheet of metal is reinforced by some kind of geodetic construction. The geodetic-sheet combination could be cast in one and then rolled to the required curvature.

It is possible that many new and perhaps novel methods of manufacture will be used to construct the large type of rocket. This will be especially so if new types of construction are used. The possible use of sandwich construction in which the core material would act as an insulator as well as the binding material for the metal sheets must not be overlooked.

The strength-to-weight factors of the heavy structural members might be improved by the use of light metal core fillings. So far as I know this method has not yet been used; briefly, the general idea and its possible advantages are these. Usually the high strength metals are relatively speaking on the heavy side. This is sometimes overcome by using thin walled hollow sections such

as tubes, etc., but unfortunately, as we reduce the thickness of the walls, elastic instability enters the picture with its consequential loss of strength and rigidity. To counteract this, the section could be filled with a core of light metal which need not have exceptional strength properties.

The metal core would not only share some of the load, but would also give local support to the thin wall of the section, thereby delaying the onset of instability. Such a section would be very much stronger than its thick walled counterpart and might possibly be even lighter. Magnesium could be used, although lithium might be more suitable.

Structural Testing

My final words will be concerned with the possibilities of testing the rocket structure as a complete unit in some kind of test rig. In order to test the structure thoroughly, it will be necessary to assimilate the actual loading which will be anticipated under flight conditions.

In order to apply the correct end-loads, the complete structure would have to be mounted in the vertical position, and point loads applied to various parts of the longerons, and possibly the tanks as well.

The tank pressure distribution might be obtained by placing the complete rocket and test rig in a deep pit filled with water. The tank units would also contain a liquid and it might be so arranged that the differences in the specific gravities of the two fluids would give, as near as possible, the correct pressure differential along the tank structure. The additional pressure could be made up by pressurizing the tanks. In this way the complete load system could be applied simultaneously to the overall rocket structure.

From these considerations it would appear that there are no insuperable problems in the design of an expendable-tank rocket. The type of construction we envisage for an initial orbital rocket will be seen in the final paper.

REFERENCES

(1) K. W. Gatland, "Expendable Rockets," *J.B.I.S.*, **7**, 160–169, July, 1948.
(2) K. W. Gatland, A. E. Dixon and A. M. Kunesch, "Initial Objectives in Astronautics," *J.B.I.S.*, **9**, 155–178, July, 1950.

III. CONCEPTION OF AN INSTRUMENT-CARRYING ORBITAL ROCKET

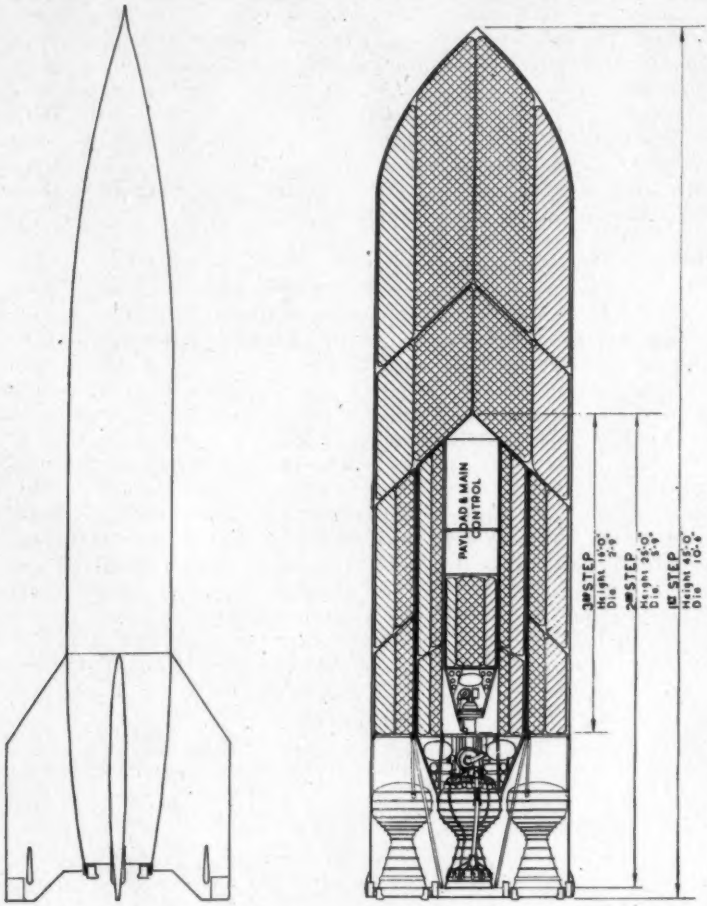
By Anthony M. Kunesch

Introduction

In the first paper this evening, Mr. Gatland gave a general outline of his thoughts on the subject of space travel generally.

In this last paper, however, I propose to cover the main design features of one particular aspect of this subject—namely, the question of orbital instrument rockets.

In our previous paper[1] we dealt with the use of expendable-tank construction as applied to a manned 5-step, Chemical and Atomic, Circum-Lunar rocket. The improvement in the performance of this project



V-2. 3-Step-expendable tank orbital rocket.

FIG. 1

made available by the use of expendable construction, led us to investigate the possibility of applying this principle to a 3-step Chemical Instrument rocket (Fig. 1) designed to reach an orbit at 500–600 miles with a 110 lb. payload. The results of this preliminary investigation were given in detail in the first part of the paper; briefly our findings were that—assuming that seven V2 rocket motors are mounted on a combined thrust-ring, each motor retaining its own feed system, and burning the same propellant combination as used in V2 or Viking, it would result in a take-off weight of approximately 135 tons with

a total thrust value for the seven motors of 190 tons at take-off (assumed at a height of 13,000 ft.), increasing to 208 tons maximum. To boost the acceleration at take-off a number of solid fuel semi-restricted booster units would be fitted to the thrust ring of the first step. The effective mass ratio of the first step, allowing for the expendable tanks, was 5·31 and with the second and third step weighing 21,610 lb. and 1,540 lb. respectively, with the effective mass ratios of 4·25 and 3·73, the overall effective mass ratio became 84, giving the characteristic velocity of 10·3 km./sec.

Without the use of the expendable tanks, the effective mass ratio of the project would be 58·5; therefore, in this particular design, the use of expendable construction provides an increase in mass ratio of approximately 40 per cent.

The potential value of expendable construction, as applied to moderate performance rockets, having shown promise, it was decided to investigate further on these lines the design problems of an instrument rocket calculated to reach an orbit at approximately 500 miles.

500 Mile Orbital Instrument Rocket

For purposes of design we had to decide on a typical propellant mixture; the main factor we considered, was that the design should be realistic, and to achieve this we should not assume the use of pure liquid ozone, or fluorine, or such fuels as liquid hydrogen and the metals beryllium, lithium, etc. Although an extremely high performance may eventually be obtained from such materials, the problems to be solved before these performances can be obtained are formidable, to say nothing of the question of handling safely as an operational project large quantities of liquid hydrogen and fluorine. A second factor to be considered was the question of propellant density, since the denser propellants give superior structural ratios:—

defined as $$\frac{\text{structural weight}}{\text{structural weight} + \text{propellant weight}}$$

and referred to as ϵ.

There are a fair number of propellant combinations that would conform to the above points. The one that has been taken for this project is liquid oxygen/hydrazine, which provides a good compromise between performance and availability. The specific gravity of the mixture of 1·05 is reasonably satisfactory on the question of density. At chamber pressures of 20 and 50 atmospheres, we would get values of 295 and 350 specific impulse sec. respectively (*in vacuo*). In practice about 95 per cent. of the figures given should be obtainable, or 280 and 330 S.I. secs. Taking a mean figure of 306 gives a value for Ve of 3 km./sec.

The general design (Fig. 1, Table I), follows the previous conception with one notable exception. The second step rocket, instead of being situated immediately above the motor bay of the first step, has been dropped down, thus doing away with the central motor of the first step. This allows the second step motor to do useful work throughout the burning time of the first step instead of lying idle over this period.

All structural requirements of tanks and supporting structure were calculated on the principles explained by Mr. Dixon in his paper on structural design.

TABLE:—THREE-STEP ORBITAL INSTRUMENT ROCKET (EXPENDABLE CONSTRUCTION)

	Overall project		First step		Second step		Third step	
	Kgs.	lbs.	Kgs.	lbs.	Kgs.	lbs.	Kgs.	lbs.
Total weight ..	77,112	170,000	77,112	170,000	10,523	23,200	817	1,800
Payload weight ..	159	350	10,523	23,200	817	1,800	159	350
Propellant weight			53,566	118,090	7,140	15,740	499	1,100
Structure weight (includes control instruments) ..			13,023	28,710	2,567	5,660	159	350
Structural ratio ϵ			0·196		0·265		0·242	
Height (metres or feet)	13·72	45·0	13·72	45·0	7·62	25·0	4·57	15·0
Diameter (metres or feet)	3·2	10·5	3·2	10·5	1·75	5·75	0·84	2·75
Number of tank bays			3 expendable 1 fixed		1 expendable 1 fixed		1 fixed	
Mass ratio (actual)	26·1		3·27		3·11		2·57	
Mass ratio (effective) ..	32·5		3·56		3·56		2·57	
Thrust*			124,740 to 139,709	275,000 to 308,000	19,958	44,000	907	2,000
Firing time (seconds) ..	382		113		106		163	

* Thrust figures for first step are at sea level, and *in vacuo*.

TABLE 1

With the required orbital height fixed at 500 miles, after making due allowance for the retardation of gravity and air drag, and losses incurred during the starting and stopping of the motors, etc., we found that the characteristic velocity required was 10·1 km./sec., for the final design. With the value of Ve at 3 km./sec., the mass ratio necessary to reach this velocity was 32·5, after making due allowance for the quantity of propellant that is required per second to provide the motive power for the pump system.

A number of papers in the last few years, and in particular, Malina and Summerfields's "Problem of Escape from Earth by Rocket,"[2] have given figures for the all-up weights of rockets consisting of five or more steps, designed to carry a payload of the order of 10 lb., to escape velocity. The basic payload around which this project has been designed, however, was fixed at 350 lb.

Since the third step is firing tangentially to the Earth's surface it is undergoing no "g" loss, a low acceleration may be used, giving the dual advantage of reduced motor weight, and more sensitive control of velocity towards

the "all-burnt" stage. The thrust of this step has therefore been taken at 2,000 lb., with a firing time of 163 seconds and a maximum acceleration of 2·9 "g." No expendable-tanks have been considered in this step due to the layout. This is, of course, dictated by the desirability of placing control instruments and payload (which in all probability would also contain sensitive instruments), as far as possible from the heat and vibration caused by the rocket motor. The height allowed for the payload and control compartments is 7 ft. 6 in.

Steering of the second and third steps of this project would be by pivoted motors controlling pitch and yaw, with roll correction by steam jets from the exhaust of the turbine in the propellant pump unit. The steam generator of the second step would continue to supply steam during the period of free-flight between "all-burnt" condition of the second step and the start of firing of the final orbital step. This steam would be exhausted through jets to maintain correct alignment of the second and third steps and would thus ensure that the third step will be in the correct attitude when it starts firing.

The first-step (the payload of which consists of steps Nos. 2 and 3 with a weight of 23,200 lb.), has been designed with 3-expendable and one fixed tank bay. The first two tank bays are situated above the second step and therefore occupy the entire cross-section of the hull. The two lower tank-bays, however, are surrounding the second step and must, therefore, be annular.

In this design the first step has six motors surrounding the motor of the second step which, as previously stated, functions as a *first-step* motor for half of its total period of burning. The steering of the first step, we consider, would be by means of exhaust vanes as in V2, due to the close multiple mounting of the seven motors. This arrangement would probably necessitate the locking of the steering mechanism of the second-step motor until separation, which would occur 113 seconds after take-off.

The propellant feed would require a self contained pump unit mounted above each motor, whilst the propellant feed for the second step-motor, during the period it is functioning as part of the first step, would be taken from the tanks of the first step via a low pressure boost pump through feed pipes and a break-away non-return valve and thence into the propellant tanks of the second step, which are thus kept topped up until separation.

Structural Considerations

The previous speaker has given you a general summing up of the main structural problems that will occur in the use of expendable-construction, and, since the design I have just detailed may be considered a typical future application of this principle, it may be of interest to picture these conclusions in a practical application.

We will start by considering the loading conditions in the tank structure of the first-step during take-off. At that moment the actual acceleration experienced amounts to approximately 1·7 "g"; this with full tanks and mean fuel density of 1·05, gives us a pressure loading varying from zero at the top of the nose tank to a maximum of 28 lb./inch^2 at the base of the bottom tank. To these pressures must be added the pumping pressure of 1·3 atmospheres or

20 lb./inch.^{2} The total pressure acting at the base of the nose tank is 34 lb./inch^2, and with internal division walls, insulation, longerons, transfer valves, etc., the total weight for this tank is 3,240 lb., with a value for ϵ of 0·064.

The bottom fixed tank has to cope with the maximum pressure of 48 lb./inch^2; this gives a weight of 1,520 lb., for the external skin, longerons, insulation, and internal division wall. Both bottom tanks being annular, we have to consider the inner wall which has to withstand compressive loading due to the pressure of 48 lb./inch^2. This inner wall, the diameter of which is 72 inches, requires structure amounting to a total thickness of 1·29 inches, the weight of which is 2,160 lb. making the total weight for this tank 3,680 lb., and a value for ϵ of 0·197.

An allowance of 1,700 lb. has been made for the additional structure that would have to be incorporated in the bottom tank to transmit the thrust loading into the structure and to afford support for the second-step. This structure would take the form of a series of radial baffles.

A further allowance of 2,150 lb. has been made for the main longerons that reach from the thrust ring at the base of the fuel tanks to approximately half-way along the nose tank. The function of these longerons is to resist the large bending moments incurred during corrections of pitch and yaw during the initial period of flight.

We now come to consider the tank structure of the second step, which has a diameter of 69 inches and a length for the propellant tanks of 180 inches. This step has to withstand the acceleration loads caused by the 7·25 "g" acceleration of the first step whilst the propellant tanks are full. The static pressure due to this acceleration amounts to 51 lb./inch^2 at the base of the bottom tank, to which must be added the pumping pressure, which, for the second step has been taken at 25 lb./inch^2.

At the base of the first tank the pressure is 62·5 lb./inch^2; the outer wall, longerons, insulations, and internal division wall would weigh 895 lb. The inner wall of the top tank would require structure amounting to ·70 inch thick, with a weight of 1,125 lb., giving a total weight of 2,020 lb. for this tank. The value for ϵ is 0·155.

The bottom tank has to withstand a total pressure of 76 lb./inch^2 and the inner wall structure is represented by a skin thickness of ·84 inches. The remaining structure of this tank including baffles for supporting the third step, longerons, etc., amounts to 400 lb., and the total weight becomes 980 lb.

Variation of Exhaust Velocity and Propellant Density

The effect of variations in exhaust velocity on the mass-ratio and payload of any specific project is made abundantly clear by considering a 10 per cent. increase or decrease in exhaust velocity. If, assuming the same take-off weight of 76 tons, we have an exhaust velocity of only 2·7 km./sec., we would require a mass-ratio of 48 and the payload would be reduced from 350 lb. to 116 lb. For a 10 per cent. increase, however, the mass-ratio would be reduced to 23·3 and the payload increased to approximately 620 lb. A further increase to 3·6 km./sec., would give a payload of approximately 1,000 lb., providing the propellant has a density of approximately 1·0.

This last point can be illustrated if we consider the case of the substitution of hydrogen for the fuel in the 76 ton rocket, and maintaining similar proportions, but increasing the exhaust velocity to 4 km./sec. The mean propellant density is reduced to 0·425 and with the same capacity tanks the propellant weights would be approximately 430 lb., for the third step, with an all-up weight of 1,130 lb. For the second step the propellant weight would become 6,190 lb., with an all-up weight of 10,600 lb., and for the first step the propellant weight would be approximately 46,500 lb. with a take-off weight of 79,000 lb., or 35·3 tons.

The structural weight of the first step would be altered little by the substitution of the low-density fuel due to the 1·3 atmospheres pressurization. From consideration of volume-ratio, the two top tanks would be designed to hold only liquid hydrogen, the oxidant being contained, together with the remaining liquid hydrogen, in the bottom two tanks.

The maximum pressure head at 1·65 "g" for the hydrogen would be approximately 1·8 lb./inch^2 on the bottom tank, plus the pumping pressure of 20 lb./inch^2. No internal walls would be required in the nose tank, apart from the dividing walls of the half tanks and light baffles to maintain tank shape. Insulation, however, would be a more serious problem owing to the low boiling point of the liquid hydrogen and a further allowance has been made to cover this; the total weight of this tank would be 2,050 lb., against 3,240 lb., for the nose tank with dense propellant. The ratio ϵ for the hydrogen tank is 0·4 whilst the same ratio for the dense propellant tank is 0·064. With the bottom tank, however, we have the oxygen and also the necessary division walls to separate the oxygen and hydrogen. The weight of this tank is 2,900 lb., and ϵ becomes 0·21 as against 0·197, with dense propellant.

In the case of the second-step, however, a slightly greater economy is effected due to the induced head being only 3·4 lb./inch^2 instead of 51 lb./inch^2. The tank weight would be approximately 630 lb. and the ratios of ϵ are 0·47 and 0·155 respectively. The bottom tank is also required to hold the oxidant and must therefore have an internal partition. The values for ϵ become 0·185 and 0·171.

The other main point that must be considered is that of the enormous pumping power required to handle the liquid hydrogen. The total B.H.P. would be at least three times the power required for hydrazine-oxygen propellant, and therefore, although the thrust would be reduced to keep accelerations to approximately the same figures as in the previous project, thus allowing the motor weight to be reduced, the pump weight and the pump fuel flow would be more than trebled. This would more than cancel any possible saving in structural weight. A further problem is that of cooling the motor, for the two liquids of extremely low boiling points cannot be used for regenerative cooling and therefore this may involve a secondary working fluid, used partially as a regenerative coolant, and partially as a sweat coolant in the motor.

Taking these factors into consideration, we find that the exhaust velocity of 4 km./sec. is modified by the additional pump fuel flow to approximately 3·57 km./sec. The mass ratios for the first and second steps are reduced to 2·56, and with the third step ratio of 1·74, the total effective mass-ratio becomes

11·4, with a characteristic velocity of only 8·70 km./sec. If we reduce the payload to 60 lb., and utilize the weight saved as additional propellant the mass-ratio of the third step is improved to 2·6, giving an overall mass-ratio of 17·0 and a characteristic velocity of 10·1 km./sec. The structural ratios become 0·32, 0·33, and 0·26 for the first, second and third steps respectively.

Expendable or Conventional Step Rocket?

I should like now to examine briefly the relative merits of expendable construction, and the conventional step rocket as exemplified in the V2-Wac Corporal combination.

If we consider for a moment the main design that I have covered this evening (namely the 76 ton rocket), and assume that it is a conventional step rocket (i.e. without expendable tanks), we find that the second and third steps must, of necessity, be supported by the tank and longeron structure. Therefore, although the tank structure would no longer require the heavy inner wall in the lower region of the hull, this is offset by the fact that the 23,200 lb., of the second step, magnified by the 7 "g" acceleration, would necessitate an increase in the structural weight to cater for the increased end load.

This, together with the increased bending moments that would occur, would mean that the tank structure as a whole would certainly be no lighter than for expendable tank construction. In addition, another motor assembly of 1,800 lb. weight would be required to bring the thrust of the first step to the design figure. If we assume, therefore, that the structural weights would remain the same, we see that the mass-ratios of the first two steps are those of the original design, without the increase made available by the use of expendable tanks. These figures are 3·27 and 3·11 respectively, and to obtain the effective mass-ratio of 32·5 the third step must have a mass-ratio of 3·2. This can be obtained by cutting the payload to 200 lb., and increasing the propellant weight to maintain the all-up weight at 1,800 lb.

The net effect of the use of expendable construction in this project is thus to provide a 75 per cent. increase in payload for a three-step rocket of approximately 76 tons weight, at the expense of increased complexity.

Variations in the basic design (i.e. the number of steps, etc.) have a considerable bearing when we come to consider specific cases. The expendable system provides considerable flexibility in design, since with the basic framework (i.e. motors and feed systems) a considerable variation in performance is possible, simply by varying the number of sizes of tanks in either the first or second steps. Since these tanks would probably be interchangeable, they could well be fitted to power units of different performances, thereby cutting production costs. For example, reduction of the payload to 200 lb. would cut the take-off weight by approximately 50,000 lb.; this can be achieved quite readily by leaving off the nose tank.

Another factor to be considered with expendable rockets is the ease with which they may be transported and set up, the bulkiest sections of the project described being the nose tank halves (18 ft. × 5 ft. 3 in. × 10 ft. 6 in.).

The conclusions we may draw are:—

That the use of expendable construction, although complicating the

structural *design*, would not entail *heavier* structure than in conventional rockets, and therefore, would provide a considerable improvement in the effective mass-ratio.

Further, with careful design, a substantial payload of several hundred pounds could be put into an orbit for an all-up weight that is considerably less than that arrived at in the German A9-A10 project and this without the use of an extremely high performance propellant.

REFERENCES

(1) K. W. Gatland, A. E. Dixon and A. M. Kunesch, "Initial Objectives in Astronautics," *J.B.I.S.*, **9**, 155–178, July, 1950.
(2) F. J. Malina and M. Summerfield, "The Problem of Escape from the Earth by Rocket," *J. Aero. Sci.*, **14**, 471–480, August, 1947.

COMBUSTION IN THE ROCKET MOTOR

By Professor A. D. Baxter, M.Eng., M.I.Mech.E., F.R.Ae.S.

A paper read to the British Interplanetary Society in London, on March 3, 1951.

The varied technical aspects of rocket propulsion have been discussed by this Society on several occasions[1,2] and there is a growing volume of literature on the subject. I have therefore selected one aspect only and hope to amplify some of the earlier remarks on it while viewing it from a slightly different angle. The treatment will be fairly elementary and to the experts I apologize in advance.

Combustion is the heart of the rocket process as it is in all internal combustion engines. A great deal is known of some of the processes in combustion, but these have always been referred to the older types of heat engine—first, work was studied on open flames, then in furnaces, then in piston engines and later in gas turbines. A few years ago the latter had serious combustion problems,[3] and a great deal of effort was applied to their solution. We have now much the same sort of problem with rockets, only a good deal more so. Firstly, because we supply the oxygen internally and not by scooping in atmospheric air and secondly, because the rocket operates at much higher pressures and temperatures. However, much can be learned from the gas turbine and I will not apologize for frequent references to and comparisons with it.

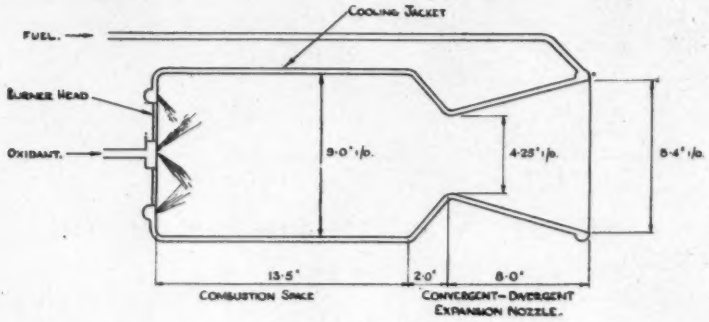
Motors may be of solid propellant, liquid mono-propellant or liquid bi-propellant type. Each has its own combustion problems and is worthy of discussion, but I propose to confine my remarks to the last type, that is, to assume that liquid oxidant and liquid fuel are injected separately.

Overall Considerations

The individual droplets of propellant start their journey through the combustion chamber *en masse* and it is probably worth while examining the overall processes before attempting to analyse the detailed component steps.

For this purpose let us consider a rocket combustion chamber giving 5,000 lb. thrust. It will be assumed that the propellants are liquid oxygen and a

hydrocarbon fuel such as petrol. Fig. 1 is a diagrammatic arrangement of such a chamber, giving the main dimensions and operating data. Other details are given in the Appendix and the whole represents conditions which might hold in practice, but which are not necessarily ideal.



PERFORMANCE DATA

$$\frac{\text{Liquid Oxygen}}{\text{Gasoline}} = \frac{2}{1} \text{ by wt.}$$

Chamber Pressure	20 atm.	Specific Impulse	219 sec.
Chamber Temperature	2830° C.	Propellant Flow	22·9 lb./sec.

FIG. 1. Layout of 5,000 lb. thrust combustion chamber.

The fuel is fed into a cooling jacket around the chamber at the expansion nozzle exit and flows towards the chamber head. At the head, the fuel is led into an injector and thence to the combustion space where it meets the oxidant which is injected directly. Here the two propellants mix and burn, forming high temperature gas which passes out through the convergent-divergent nozzle. The latter reduces the gas pressure and increases its velocity.

The main processes within the chamber can be listed under three heads:

(1) injection;

(2) ignition;

(3) combustion.

These are not entirely disconnected processes and cannot be completely separated in the motor. They do form, however, a convenient division for discussion.

Injection

Gas turbine experience has shown that atomisation of the fuel is important because the rate of combustion depends upon the exposed fuel surface. The surface associated with any given volume is increased by sub-division. This can be shown mathematically and, in fact, the surface increases, in the case of spherical droplets, inversely as their radius. The number of droplets, however, increases inversely with the cube of the radius. In practice, injectors are designed to produce droplets varying between 0·001 in. and 0·010 in.

diameter (25μ to 250μ). Thus a thimbleful of fuel may be broken up into perhaps six million droplets whose surface area will have increased from about that of a couple of large postage stamps to the equivalent of the internal wall surface of the 5,000 lb. thrust chamber. At the same time it should be borne in mind that some three thousand thimbles full will be injected each second. Consequently we can soon reach astronomical figures within the chamber and the individual droplets fade into insignificance.

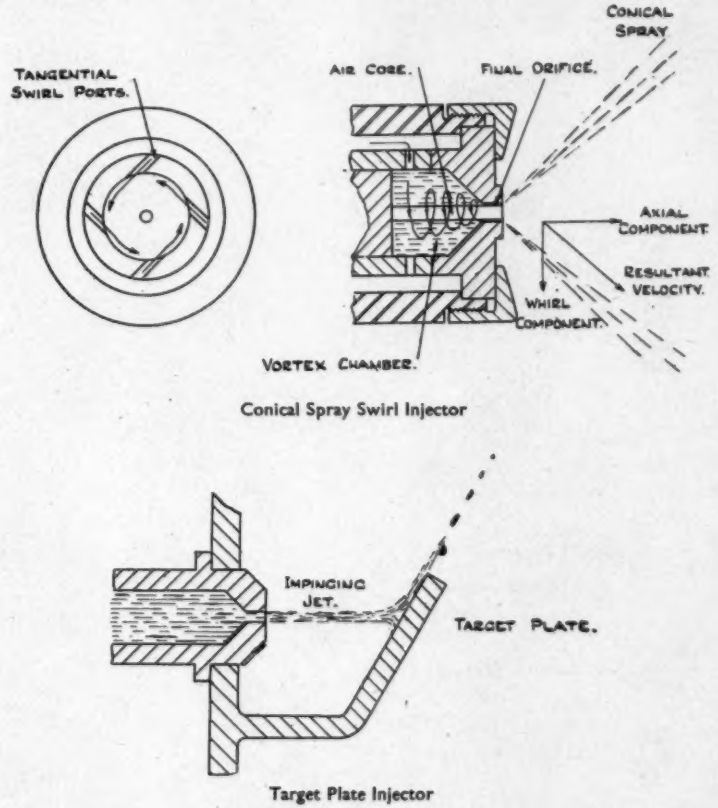


FIG. 2. Conventional atomizing injectors.

This multiplication of droplets is not achieved without the expenditure of energy and much work has been carried out in various fields on the problem of atomisation.[4] Most injectors rely on either internal turbulence or external impact of the fluid to produce atomisation. Examples applied to rockets

are the conventional swirl jet and the "target plate" type which are shown diagrammatically in Fig. 2. The former consists of a body with an inner part containing a set of entry holes leading tangentially into a small swirl chamber and thence through an exit orifice. Fluid is fed into the injector and is given a rotary velocity by the combination of tangential holes and the swirl chamber. As it moves towards the exit, it is forced inwards and its

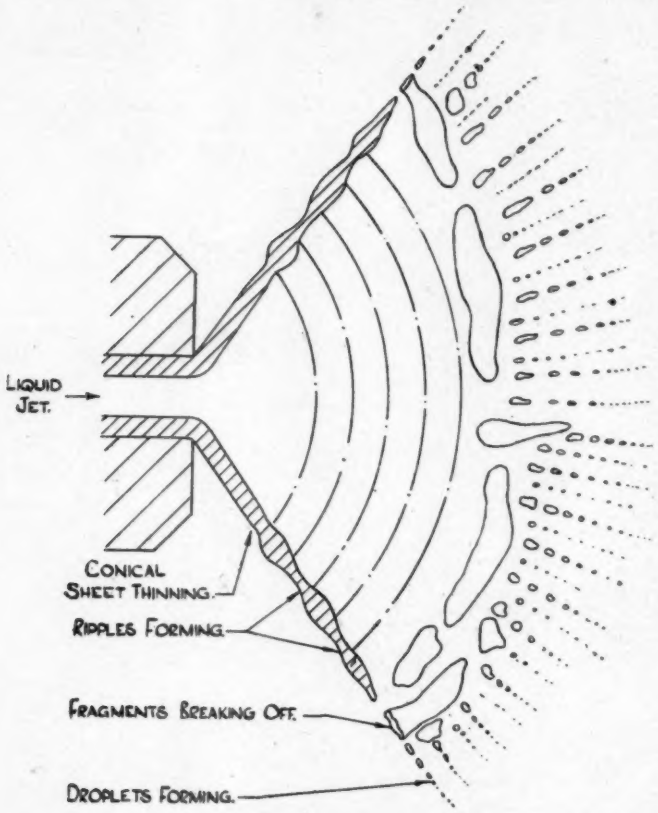


FIG. 3. Mechanism of atomization.

tangential velocity increases inversely with the radius. Leaving the final orifice, all constraint on the fluid is removed while it has both an axial velocity component and a centrifugal one. The combination of these produces a hollow cone in which each drop follows a straight path.

Target plate injectors consist essentially of plain circular or slit orifices which permit the fluid to impinge on a surface. At impact the direction of

flow is changed and the fluid spreads itself in a thin uniform sheet over the plate. It leaves the plate with this uniform distribution and the atomisation is, thereafter, similar to the swirler type. The break-up mechanism is shown in Fig. 3. As the cone spreads out the film thickness becomes less in order to maintain continuity of flow. At the same time the changes in direction of the fluid flow produce internal turbulence which becomes evident as the film becomes thinner. Ripples are formed and irregular thinning occurs at intermediate points until surface tension is no longer sufficient to prevent disintegration into fragments. These fragments then continue the break up process into cords and strings from which the droplets are born. Some very fine photographic studies of conical spray formation have been made at the Royal Aircraft Establishment[3] and elsewhere[5] illustrating this process.

The foregoing is not the whole mechanism as the liquid enters the chamber with a high initial velocity relative to the surrounding gases. For an injection pressure drop of 100 lb./sq. in., the velocity will be of the order of 130 ft./sec. The gases exert friction forces upon the film which tend to slow it down. At the same time these assist the break up by causing more turbulence and uneven conditions. The whole process varies with Reynolds Number, and high injection pressure and low viscosity will speed up and improve atomisation. Once the droplets have been formed they will rapidly be reduced to rest relative to the surrounding atmosphere except for very large drops. High chamber pressure will assist in this direction. The net result is that the penetration of the spray is not great and the whole process in a chamber working at 300 lb./sq. in. would occupy a chamber length of a little over one or two inches and a time interval of approximately one milli-second.

The second objective of the injector is to ensure good mixture distribution. In the case of the gas turbine liquid fuel is injected into more or less uniformly distributed air. Thus the injector has to spread the fuel evenly through the air so that each droplet is completely surrounded by its correct ration of air (or oxygen) to give complete combustion. In the rocket there are two liquids to consider and the problem is how to surround the fuel by its ration of oxygen when the latter is itself contained in a liquid droplet. It might be suggested that atomising the oxidant to a much finer degree than the fuel would provide the answer. Each fuel drop would then be surrounded with oxidant drops, but this would mean expending a great deal of extra energy on atomising the oxidant; alternatively the fuel could be left in relatively coarse droplets. Neither would be very attractive. Fortunately it usually happens that the volume of liquid oxidant is several times that of the fuel and a break-up into equal sized fuel and oxidant droplets will give the possibility of some measure of satisfactory distribution.

These, then, may be the ideal conditions, but can they be achieved? The answer is that, in most cases, practice still falls far short of the ideal. The effort to produce this ideal has resulted in much ingenuity and some peculiar designs of burner head, as the complete combination of propellant injectors is termed. It is hardly necessary to give any detailed description here as published reports are available.[2,6,7etc.] An important objective in the designs should be to prevent the droplets having a high axial velocity, otherwise useful

combustion space will be wasted before the combustion reaction begins. On the other hand, too great a radial velocity may result in impingement on the walls and too low an axial velocity may encourage combustion too close to the burner head with consequent overheating. In the case of the gas turbine, the axial velocity is not quite so important as the air flow is arranged to give a recirculation effect near the axis which is of great benefit in the early stages of combustion. No efforts are made to produce reverse flow in the rocket and as indicated above, it might actually be dangerous.

Some small degree of re-circulation will, however, be produced by the entrainment of gas with the spray as it moves forward. This will give substantial help in transferring heat to the propellants which will, in addition, be subjected to intense radiation from the flame further downstream. Together these will help to prepare the propellants for the next phase in their brief life within the chamber. Because of the high chamber pressure, the droplets will be near their critical pressure; for example, kerosene has a critical pressure of 250 lb./sq. in. and critical temperature of 400° C. It has been shown that in these conditions[8] the heat transfer rate is many times greater than normal. Thus, evaporation will proceed and with some of the smaller droplets be completed. With larger droplets still moving relative to the chamber atmosphere, a trail of vapour will be left behind. Adjacent layers of fuel and oxidant vapour will diffuse and form the first centres of combustible mixture. At this point it can be assumed that the injection phase is complete and combustion begins.

Ignition

When the droplets of fuel and oxidant have been injected into the combustion chamber and have been mixed as carefully as can be arranged, the next thing is to secure their ignition as promptly as possible.

This phase in their life is of vital importance, for if ignition is delayed even briefly, enough propellant can collect to cause a "hard" start or possibly quite a violent explosion. This can be illustrated with our example 5,000 lb. motor. Its chamber volume is approximately 850 cu. ins. To fill this volume with gas at the working pressure of 20 atm. would require the reaction of 0·08 lb of propellant which, at the normal rate, would be injected in 0·0035 sec. If 1·0 lb. of mixture collected and then ignited instantaneously an explosion would occur which would give a chamber pressure of 250 atm. if the chamber could withstand it. With atmospheric pressure conditions prevailing in the chamber before ignition, the propellant flow could, unless special precautions were taken, rise to nearly double its normal rate and 1 lb. be injected in 1/40th sec. Special precautions are usually taken and consist, sometimes of starting with a low injection pressure and sometimes in using only a proportion of the injectors. These are then co-ordinated with the method of ignition.

Ignition may be by

(*a*) external heat energy supply;

(*b*) self-igniting propellants.

The first can take a number of forms as, for example, sparking plug, electric glow plug, pilot flame or pyrotechnic. The method selected depends upon various considerations, one of which is whether ignition will be required more than once during a single flight. If it is, pyrotechnic igniters may not, amongst others, be very convenient.

It will be shown in the next section on combustion that continuous ignition is maintained by the supply of heat and activated gases to the unburnt propellants. The heat energy required for this purpose is only a fraction of the total released in full combustion, but unless that fraction is achieved the reaction rate will slow down and may result in flame extinction. In the case of single droplets, the heat required is that which will raise them to their boiling point, vaporise them and raise the vapour to the self-ignition temperature. The heat generated in the subsequent combustion is then used to put adjacent droplets through the same cycle, continuing until a fully established flame front is formed.

The heat necessary to start this cycle with liquid oxygen and a single droplet of kerosene (100 μ dia.) would be approximately 0·05 calorie while the potential heat release would be 0·65 calorie. This demonstrates that the theoretical heat needed for ignition is very small, but in practice a number of factors intervene to necessitate an increase. First, in some propellant combinations an endothermic decomposition of the droplets may be a preliminary to ignition. Nitric acid, for example, probably breaks down along some lines such as the following:—

$$2\ HNO_3 + \text{heat} = H_2O + 2\ NO_2 + \tfrac{1}{2}O_2$$

Next, it has been assumed that all the energy goes into a single pair of droplets which produce a mixture neither too rich nor too weak for ignition. Finally, it is assumed that the droplets, on ignition, do not themselves lose heat energy to more than the amount of propellant which can be brought to the ignition temperature.

If ignition did occur from one droplet, the burning zone around it would be a very small radius sphere. This would lose heat at a high rate and the droplets in close proximity might then be heated insufficiently to ignite. In practice, the ignition source will be surrounded by a large number of droplets and it must supply sufficient energy to ensure that quite a number reach the correct mixture conditions and ignition temperature.

Ignition occurs after a "delay" period during which pre-flame reactions are occurring but, as pointed out above, this delay must not be great. In practice delays of more than 0·010 sec. cannot normally be tolerated. It is conditioned, like other combustion reactions, by pressure and temperature, increasing as they are reduced. On this account considerable heat energy in excess of the minimum is desirable for good ignition. Designers, therefore, in order to be safe, provide a temporary heat source which is sure to be sufficient. As a limit in this direction it can be assumed that all the propellant entering the chamber must be heated to the self-ignition point. This would demand an output of up to 10% of the propellant potential heat output. By reducing the initial flow rate, the igniter output could be reduced and this is frequently done. One

method is to use a pilot chamber where combustion is established with small flow rates. The hot combustion gases then pass into the main chamber where their heat is sufficient to ensure ignition of the main propellant flow.

A very brief ignition energy output period should be all that is necessary, but to be certain of safe starting, the igniter should start to function slightly before propellant injection and should be maintained until stable combustion is established. In the ideal chamber, a time of 0·02 sec. would be adequate, but in practice times of 0·5 to 1·0 sec. may be employed.

The second method of ignition by self-igniting propellants has many attractions and some drawbacks. Examples of such propellant pairs are hydrogen peroxide/hydrazine hydrate, nitric acid/aniline and liquid oxygen/zinc di-ethyl. These combinations react exothermally in the liquid state and at normal temperatures. The reaction generates heat at a rapid rate which causes the droplets to boil, forming a vapour mixture in which reaction continues to raise the temperature to the self-ignition point. Once burning has started the heat release rate increases until the steady state combustion condition exists. The first stages of this reaction may be relatively slow and a delay will exist as in other ignition methods. Under altitude (low pressure) conditions this delay may become excessive and could be serious in rockets where starting at extreme heights is contemplated.

During the delay, the propellants will move towards the exit of the chamber. If ignition occurs near the exit, a flame front will build up there, pressure and temperature upstream will rise and the flame will move forward as the reaction rate of the propellant increases until an equilibrium position is reached. The forward movement of the flame may be sufficiently rapid to give the "hard start" feature already mentioned. In this connection chambers of large volume/propellant throughput ratios are less sensitive.

At this stage it can be assumed that the dangers associated with ignition have been overcome and that propellants are burning in a stable state. We can therefore proceed to an examination of the combustion phase.

Combustion

Before considering the fate of individual droplets, let us again survey the overall scene in our combustion chamber. Propellants are atomising, evaporating and reacting at enormous speeds. In one second 2¼ gallons of liquid have entered the chamber and been converted to high temperature, high pressure gas occupying a volume nearly three hundred times as great as the chamber volume. From this it may be deduced that the average "time of stay" of the propellants in the chamber is 3·5 milli-seconds.

The heat released during the chemical reaction raises the gas temperature to nearly 3,000° C. and the actual quantity of heat would be sufficient to provide district heating for a town almost as large as Brighton! Stated more precisely the combustion intensity is 20×10^6 C.H.U./cu. ft./atm./hr. This compares with gas turbine intensities of about 2×10^6 C.H.U./cu. ft./atm./hr.

The chemical reaction, written in generalised form, is

$$a.\ C_nH_m + b.\ O_2 = c,\ CO_2 + d.\ H_2O.$$

Before reaching the state of complete combustion indicated on the right hand side of the equation, much intermediate reaction has to go on and, paradoxically, because of the high temperature resulting from the combustion, the reaction does not get beyond some of these intermediate stages. This is due to dissociation, which allows gases such as CO, H_2 and O_2 to remain in the same mixture without reacting together. The first two are produced from the original fuel and where they exist, imply that all the chemical energy of the fuel has not been converted to heat. Hence the final gas temperature will be less than might have been expected. This can be allowed for and is, apart from the tedious calculations involved, regarded almost as a blessing by the practical engineer who has to cool the combustion chamber.

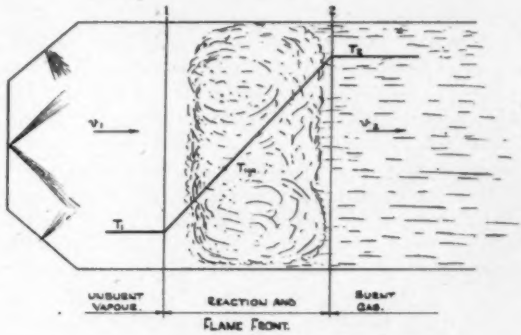


FIG. 4. Ideal combustion reaction.

Again considering the ideal process, we can assume that a flame front exists across the chamber as drawn in Fig. 4. Ahead of it propellant mixture moves towards it with velocity v_1 and temperature T_1. Behind it, the reaction is complete and burnt gases at temperature T_2 move downstream with an increased velocity v_2. Between planes 1 and 2 the whole reaction occurs. The mixture must be raised to a temperature $T_{ign.}$ where burning starts. The heat of reaction will then be released and will raise the gas temperature further until it finally reaches T_2 and completes combustion at plane 2. At the same time some heat will be conducted back to the unheated mixture arriving at plane 1 and will raise its temperature to $T_{ign.}$, when the reaction will continue. This is a simplified picture and is not complete as diffusion of hot gases and intermediate products play an important part in promoting combustion in the fresh mixture.

Between T_1 and $T_{ign.}$ droplets will be evaporating and starting on the preliminary stages of reaction. This is the "delay" period, corresponding to that discussed in the section on ignition but very much shorter because of the much higher temperature and pressure conditions prevailing with continuous combustion. This delay in hydro-carbon reactions is associated with the formation of aldehydes and peroxides which then have an accelerating effect on combustion. The normal "flame front" (planes 1 to 2) in which all this

occurs is theoretically only a minute fraction of an inch in thickness—about 0·0005 in. at atmospheric pressure and much less at chamber pressures. Actually, because the mixture is not homogeneous and more so because some of the propellant is still liquid, combustion will continue beyond plane 2.

Consider now the velocities before and after combustion. The flame front is stationary in the chamber because its speed towards the burner head is equal and opposite to the speed of the mixture feeding it. Behind the flame front, the hot gases will have accelerated to a much higher speed $v_2 = v_1 \cdot \frac{T_2}{T_1}$ approximately. Any unburnt propellant will be swept along at this speed and hence the burning, instead of being confined to the very narrow flame front, is spread over a large section, and it often happens that combustion is not complete even at exit from the chamber.

It was indicated at the beginning of this section that the average "time of stay" of a droplet in the chamber would be 0·0035 sec. if all the propellant burnt instantaneously immediately after injection. As shown above, this does not occur and as a result the droplet life may be modified quite considerably. If, for example, the velocity of the burnt gases through the chamber when complete combustion occurs is 400 ft./sec., the time of stay in a chamber of 1 ft. length from the flame front would be 0·0025 sec. If only nine-tenths of the propellant burnt at the original flame front, the remainder would be swept along with the hot gases at 360 ft./sec. and would have up to 0·0028 sec. to complete its combustion. Similarly, if only one tenth burned, the remainder would be carried through the chamber at only 40 ft./sec. and would not reach the end until 0·025 sec. later. Thus quite an appreciable period for evaporation and mixing can elapse. In practice it is probable that the burning rate starts slowly, speeds up as droplets evaporate and mix within the initial hot products and finally slows down as the last scattered droplets seek the partners they need for combustion. The process is illustrated graphically in Fig. 5. The upper diagram shows the gas velocity through the chamber for the practical and ideal cases and the lower diagram gives the life time of a droplet. The shading in the combustion chamber indicates the amount of burnt gas at any point in the chamber.

It will be noted that in the hypothetical practical case the total available burning time is four times as long. Assuming a small unburnt fraction in the other case, to obtain the same time for it to burn as the last fraction in the practical example would require a chamber of four times the length. If, of course, there had been no unburnt fraction, the chamber could have been greatly reduced in length. This length L in cylindrical chambers is related to the design parameter L* by the formula

$$L^* = \frac{A}{A_t} \cdot L$$

where

A = cross section area of the chamber

A_t = cross section area of the venturi throat.

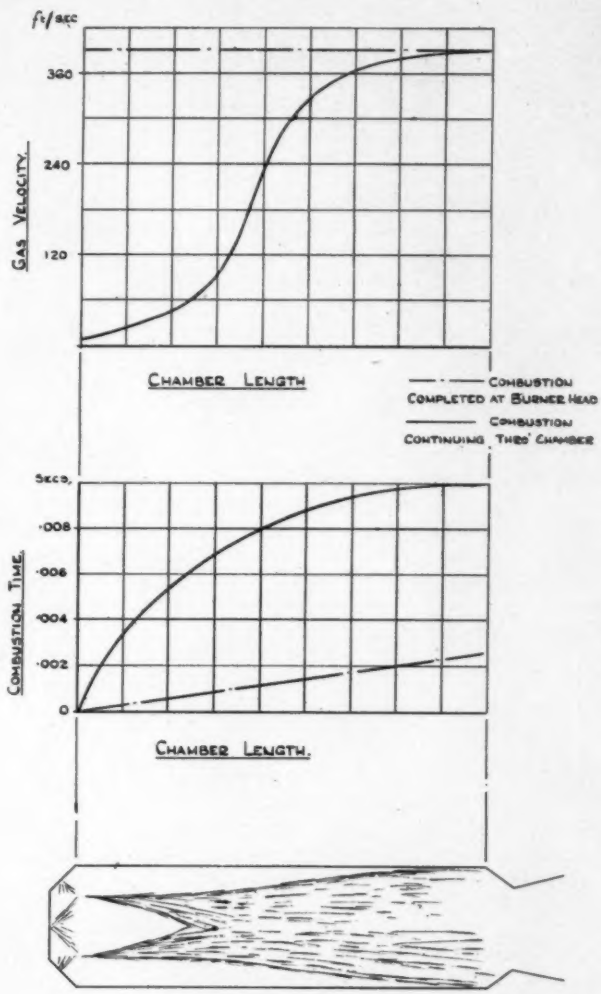


FIG. 5. "Time of stay" diagram.

A previous lecture[2] has indicated how performance varies with L* or, in our case, with L. A drop in thrust is experienced as the length is reduced from an optimum value because combustion is not completed. Unburnt propellant reduces the gas velocity so that the completeness of combustion can be taken

as proportional to the gas velocity in the chamber. Hence at all chamber lengths less than that giving complete combustion, the chamber having the slower burning rate will give a worse performance than the other. Thus a burner head giving rapid combustion is worth striving for.

The "normal" flame velocity of hydrocarbons burning in air is only about 1 ft./sec.; in pure oxygen it is somewhat higher and it will, of course, be affected by pressure and temperature. Nevertheless, the propellant velocity v_1 will, in general, be much higher. If the propellant were all vaporized it would have an average forward speed of some 20 ft./sec., but the droplets will have only partly evaporated and the speed will be less. However, for the flame front to counter this, it is obvious that it must have recourse to some re-circulation and stabilizing flow conditions such as hold in the gas turbine and were suggested in the section on injection.

Turn now to the combustion of individual droplets. First, it should be made quite clear that direct burning of the liquid fuel does not take place at the surface as occurs with solid fuel, but all reaction is in the vapour phase. Thus the liquid droplet evaporates and is surrounded by a layer of vapour. When this vapour meets oxygen vapour in suitable proportions it will burn. In the ideal case, therefore, each droplet of fuel would be surrounded by a thin spherical flame front held away from the liquid by the radially outward movement of the vaporized fuel and supplied with oxygen vapour from the outside. There is, of course, no reason why the converse should not occur with the oxidant droplet in the centre. If, for example, instead of liquid oxygen which evaporates at low temperature, nitric acid were used, it is quite conceivable that a light hydrocarbon fuel such as petrol, would be vaporised first and so surround the liquid acid.

In both cases complex conditions of heat and matter transfer occur. These have been studied in the laboratory[9,10] for the fuel in air case and theories of combustion developed. The rocket case is more complicated, but can be described as follows. Fuel and oxidant droplets are adjacent and under the influence of heat they begin to evaporate from their surfaces. The vapours will diffuse towards each other, eventually forming a combustible mixture zone. When combustion occurs the heat released will raise the temperature of the products to T_r, but some heat will be conducted back to the liquid droplets at temperatures T_f and T_o by the temperature gradients across the diffusing vapours. This must be sufficient to supply the latent heat of vaporisation and may, if greater, raise the temperature of the liquids.

Thus, the burning rate of droplets is effectively their rate of evaporation. This depends, as shown, on the rate of heat flow to the droplet and upon the heat required to raise its surface to the boiling point and supply latent heat of evaporation. From this it can be deduced that droplets burn at a rate proportional to their diameters and that the time for complete combustion is proportional to the square of the initial diameter.

At atmospheric pressure, a droplet of 100 μ (0·004 in.) diameter would burn in still air in approximately 0·010 sec. Within a rocket combustion chamber, however, it would be subject to very high temperature and pressure and it is reasonable to suppose that both would have some influence in reducing the

burning time. Little has been published on this subject and any figures given can be no more than estimates. It would seem that the various factors might produce between a three and fourfold reduction in burning time and that the actual time would be about 0·003 secs.

From the earliest analysis of possible "times of stay," this figure would suggest that the maximum size of droplet that could be expected to burn completely within the chamber would be about 180 μ. This does not take account of any evaporation during the injection stage nor the possibility of longer time of stay than 10 milli-sec.

By the time the droplets have reached the end of the combustion chamber, they may have

(i) vaporized completely, been mixed with their partner for combustion and have burned;

(ii) partly vaporized with combustion still occurring;

(iii) partly vaporized or cracked and remained unburned.

In any event, their time within the motor is drawing rapidly to a close. They are accelerated towards the nozzle throat with a fall in gas pressure and temperature. The time required depends slightly upon the shape of the convergent section, but as the velocity changes from 400 ft./sec. to 3,500 ft./sec. in a distance of approximately two or three inches, the time will be about 0·0001 sec. From the throat to the exit, the divergent nozzle allows the gas to expand with pressure falling to atmospheric and temperature to about 1,800° C. The exit velocity would be about 7,000 ft./sec. and the time to traverse the nozzle about 0·00013 sec.

There is, therefore, very little time for the uncompleted combustion of (ii) and (iii) to proceed further, but what may happen is that some of the dissociated gas found in the chamber may re-combine as the temperature falls. The speed of this reaction depends upon temperature and pressure and as these fall the rate of re-combination will fall also. The re-combination will produce heat and consequently the final temperature will be higher than if the reaction equilibrium at the beginning of expansion had been "frozen." This extra heat, perhaps 50% of that originally lost, unless allowed for by the designer, may prove very embarrassing from the nozzle cooling aspect.

As the gases finally leave the exhaust cone, any unburnt fuel near the jet boundaries will come in contact with atmospheric air and proceed to burn externally, producing a long torch flame as shown in Fig. 6. This will occur if the fuel is unburned simply because of poor mixing, or if it has been injected for film cooling of the chamber walls, or if it is part of a rich mixture deliberately used to keep the combustion temperature down. In any case this combustion is of no value to the motor performance.

One other common feature of the exhaust jet is the appearance of standing waves which are marked by bright luminous patches. A good example of these is shown in Fig. 7. They are produced by the supersonic jet producing

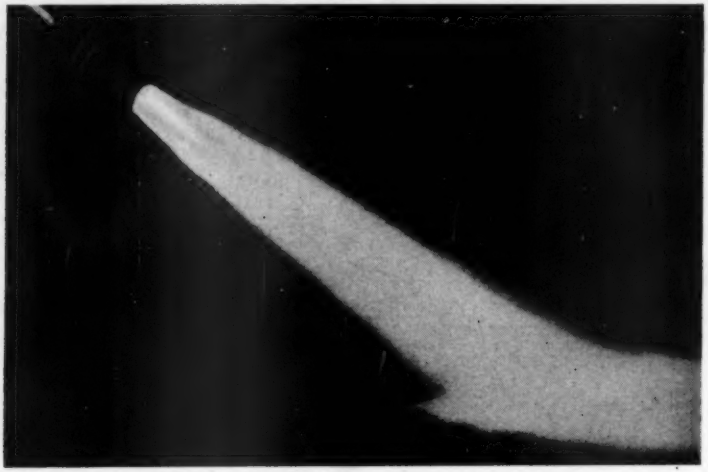


FIG. 6. Exhaust jet showing after-burning.

FIG. 7. Exhaust jet shock waves.

shock waves. A simplified explanation of them is given in Fig. 8. If the gas pressure within the jet is not precisely the same as the atmospheric pressure a sudden expansion or contraction will occur as soon as the nozzle no longer constrains the jet. This causes the gas particles to accelerate radially and owing to their inertia they tend to move beyond their equilibrium position.

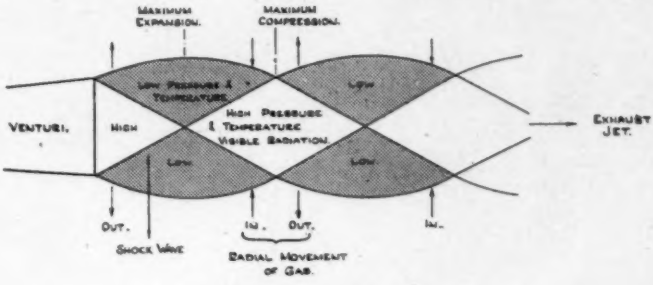


FIG. 8. Diagram of supersonic jet shock waves.

This produces a reversal in pressure condition and a periodic oscillation of the gas is set up with alternate high and low pressure areas within the core of the jet. These cause temperature changes which modify the dissociation equilibrium and excite the gas molecules so that visible radiation occurs within the high pressure regions. These are the well-known shock diamonds so well illustrated in Fig. 7.

Conclusion

The preceding sections have not fully analyzed all the rapid changes which occur in the life of a droplet as it passes through the rocket combustion chamber; in fact, they have been sketched in barest outline and many alternative life histories have been left almost unmentioned as, for example, some of the chemical changes which a droplet may undergo during combustion, the impingement of droplets on the chamber walls, the use of fuel for film cooling and so on.

Most successful chambers to date have been developed by experience, empiricism and hard trial and error. This state will remain as long as the internal processes defy comprehensive analysis. Much research is required to rectify this and to say more than that the life of a droplet is short and fiery.

Acknowledgment

Thanks are due to the Chief Scientist of the Ministry of Supply for permission to publish this paper. Figs 6 and 7 are Crown Copyright and are reproduced with the permission of the Controller of H.M. Stationery Office.

I also wish to state that the views expressed are my own and do not necessarily reflect those of the Ministry of Supply or any other Government department.

APPENDIX

Rocket Motor Design and Performance Data

Specification

Thrust	5,000 lb.	
Propellants	Liquid oxygen Hydrocarbon, 85% C 15% H	} Weight ratio = 2 : 1
Chamber Pressure	20 atmospheres	

	Theoretical	*Practical* (87% *theoretical*)
Overall Performance		
Specific impulse	253 secs.	219 secs.
Propellant flow	19·7 lb./sec.	22·9 lb./sec.
Internal Performance		
Combustion gas temperature ..	3,320° K.	3,100° K.
Throat gas temperature	3,040° K.	2,890° K.
Exit gas temperature	1,890° K.	2,060° K.

Main Dimensions

Combustion chamber L*	=	60 in.
Combustion chamber $\frac{L}{D}$	=	1·5 in.
Throat diameter	=	4·25 in.
Chamber diameter	=	9·0 in.
Exit diameter	=	8·4 in.
Chamber length	=	13·5 in.

REFERENCES

(1) J. Humphries, "Problems in Rocket Development," *J.B.I.S.*, **6**, No. 4, March, 1947.
(2) J. Humphries, "The Design of Rocket Motors," *J.B.I.S.*, **8**, No. 3, May, 1949.
(3) P. Lloyd, "Combustion in the Gas Turbine," *Inst. Mech. Eng., W.E.P.*, No. 12, 1945.
(4) D. G. Shepherd, *An Introduction to the Gas Turbine*, p. 231 *et seq.*, Constable & Co., Ltd., 1949.
(5) E. A. Watson and J. S. Clarke, "Combustion and Combustion Equipment for Aero Gas Turbines," *J. Inst. Fuel*, **21**, No. 116, 117, Oct. and Dec., 1947.
(6) W. H. Wheeler, "Rocket Propulsion," *Inst. Chem. Engrs.*, March, 1948.
(7) A. D. Baxter, "Aircraft Rocket Motors," *Aircraft Engineering*, **19**, No. 222, August, 1947.
(8) E. Schmidt, E. Eckert and V. Gugull, "Transportation of Heat in Liquids near their Critical Point," *M.A.P. Reports and Translations*, No. 95, May, 1946.
(9) H. G. Wolfhard and W. G. Parker, "Evaporation Processes in Burning Kerosene Spray," *J. Inst. Pet.*, **35**, p. 118, 1949.
(10) D. B. Spalding, "Combustion of Liquid Fuel in a Gas Stream," *Fuel*, **29**, No. 1, January, 1950.

NOTES AND NEWS

International Congress on Astronautics

The Secretary would be very pleased to hear from any members who would be willing to offer accommodation to delegates to the Second International Congress on Astronautics, to be held in London from September 3–8, 1951, together with a note of their terms.

Assistance is also sought from members who would be willing to act as interpreters during the official proceedings.

Full details of the programme are being circulated with this Journal, and particular attention is drawn to the concluding dinner which is being given by the B.I.S. in honour of the delegates at St. Ermin's Hotel, S.W.1, on September 8, 1951, at 7 (for 7.30) p.m.

Admission tickets (price £1 11s. 6d. each) are now available from the Secretary, and early application is advised.

Astronautics in Australia

We have received particulars of the 30th Anniversary Exhibition of Qantas Empire Airways, held in Sydney last year. In addition to a review of the airline's history, the exhibition gave a glimpse of the future as it included a section on astronautics. On display were an orrery, a planetarium, pictures from *The Conquest of Space*, and a space-suit from the film *Destination Moon*. A leaflet explaining the principles of space-flight to the layman was also issued, and we congratulate Qantas on their enterprise in bringing astronautics to Australia.

* * * *

From the World's Press

Sir John Cockcroft, the chief of Britain's Atomic Energy Research Establishment at Harwell, said in a recent public lecture that "one enthusiast" on his staff hoped that interplanetary flight might be achieved by the application of nuclear energy; he conceded that such dreams might be realized, if it ever became possible to employ sufficiently high temperatures in nuclear reactors. We doubt whether interplanetary interest at A.E.R.E. is confined to our Technical Director—if it is, we feel he has been doing a poor job of spreading the gospel!

* * * *

One encounters interplanetary travel in strange places nowadays: in a recent issue of *Skinner's Silk and Rayon Record* (!), we found a description of some "bright new beachwear designs" from B. L. Burton, Ltd., decorated with illustrations of Lunar landing stations, interplanetary travel bureaus, and (of course) the inevitable "space-liners" with rows of portholes and impossible mass-ratios.

* * * *

Several reviews of the book of reminiscences about Shaw by his secretary (*Thirty Years with G.B.S.*, by Blanche Patch) have seized on the revelation that he was a member of the B.I.S. Miss Patch says that G.B.S. would not have flinched from acting as our own messenger to the Moon, if our equipment had been ready for him! He apparently was firmly convinced of the universality of life, and regarded with scorn any argument seeking to "prove" the unlikelihood of extra-terrestrial beings, by comparison of conditions on other planets with those on this. He is even quoted as going so far as to say "Neither the heat of the Sun nor the cold of the Moon raise any presumption that both these bodies are not densely populated by beings adapted to their environment."

* * * *

One Separovic, commenting from Radio Zagreb (Yugoslavia) on February 25, said that a Soviet poet had taken advantage of current world interest in space-flight to write:—

> "Your august thoughts mean light to me,
> O Stalin, with them you are giving warmth to nations,
> O lighthouse of all the planets."

His own interpretation of this was that Russia would now seek to export Stalin's statements to other worlds, because people here no longer believed them.

ABSTRACTS

Edited by J. HUMPHRIES

Abbreviations of titles of journals were given in the May, 1950, issue of the *Journal* and addenda have appeared in all subsequent issues. The following is a further addendum to the list.

Ing. Arch.	Ingenieur-Archiv.
J. Geophys. Res.	Journal of Geophysical Research.
J. Roy. Aero. Soc.	Journal of the Royal Aeronautical Society.
Prikl. Mat. i Mek.	Prikladnaya Matematika i Mekhanika

AIRCRAFT

(See also abstract No. 116.)

(91) **Performance boosting of high-speed fighters with rockets.** Z. PLASKOWSKI, *Flug. u. Tech.*, (6) (1950). (*In German.*)
Considers a jet-propelled fighter with rocket boosting of 25 and 50 per cent. of the jet thrust. Comparisons of take-off distance and time, rate of climb, time to reach given height and ceiling are detailed.

(92) **Atomic-powered aircraft propulsion.** A. KALITINSKY. *Aeroplane*, **80**, 172-173, 176-177 (*9th Feb.*, 1951). Atomic energy as applied to the closed-cycle turbo-prop, the ramjet and rocket.

ASTRONAUTICS

(93) **Problems of space flight.** J. STEMMER. *Flugwehr u. Technik*, **11**, 279-284 (*Dec.*, 1949). (*In German.*) Elementary discussion of rocket propulsion and the space station.

(94) **With rockets to the Moon.** E. BIALOBORSKI. *Przekroj* (262) (1950). (*In Polish.*)

(95) **Shall we travel to the Moon one day?** E. BIALOBORSKI. *Przekroj* (277) (1950). (*In Polish.*)

(96) **The interplanetary rocket.** E. BIALOBORSKI. *Urania* (96/98), 12-21 (1950). (*In Polish.*)

(97) **Possibilities and problems of space-flight.** H. SIEDENTOPF. *Universitas*, 53-62 (1950). (*In German.*)

(98) **Signals from the Moon and Mars.** G. SCHRUTKA. *Natur u. Technik*, **4**, 271-274 (*Oct.*, 1950). (*In German.*) Considers the possibility of communicating with a space-ship by means of light-signals.

(99) **Technical questions of space-flight.** *Natur u. Technik*, **4**, 329-330 (*Dec.*, 1950). (*In German.*)

(100) **Instruments for upper atmosphere and interplanetary navigation.** J. C. BELLAMY. *Navigation*, **2**, 272-275 (*Dec.*, 1950).
Considers the limitations of present aircraft navigational instruments and possible future instrumentation of space-ships.

(101) **Fundamentals of interplanetary flight.** Pt. I. K. A. EHRICKE. *Forschungsreihe d. N.W.G.f.W. no.* 1, 22 pp. (*Dec.*, 1950). (*In German.*)

The introduction deals with basic principles of rocket motor operation and considers both chemical and atomic rockets. The problems involved in space-flight are then dealt with in detail, the various steps in achieving interplanetary flight being considered. Finally some of the limits of interplanetary flight using chemically fueled rockets are delineated.

(102) **Rocket navigation.** S. HERRICK. *Navigation*, **2**, 259-272 (*Dec.*, 1950).
Discusses first of all basic astronomical laws governing interplanetary flight. Then considers flights from Earth to Venus and Earth to Mars using minimum energy orbits, i.e. rectilinear orbits in initial and final phases with heliocentric ellipses in the intermediate phases. An appendix summarizes the necessary formulae and constants required for calculating orbits and gives several examples.

(103) **Electric space-ships.** Pt. II. H. OBERTH. *Radio-Electronics*, **22**, 74, 76, 78, 80, 82 (*Jan.*, 1951).
Details of a space-ship and its power plant using the sun's rays as energy source.

(104) **The gravity field of two bodies.** W. SCHAUB. *Weltraumfahrt*, **2**, 4-10 (*Feb.*, 1951). (*In German.*)
Deals with the effect of the Earth and Moon on a space-ship.

CHEMISTRY

(105) **Engineering materials for use with Becco 90 per cent. hydrogen peroxide.** Pt. I. Materials of construction. *Buffalo Electro-Chemical Co., Inc.*, 25 pp. (1*st Dec.*, 1950).
Deals with the determination of compatibility of engineering materials with 90 per cent. hydrogen peroxide, determination of its concentration and determination of impact sensitivity of materials containing it. Classified tables of compatible materials for short and long-term contact are given. 99·6 per cent. aluminium is recommended for long-term storage tanks whereas for short-term work many aluminium alloys and stainless steels may be used. Plastics and lubricants are also dealt with.

MISCELLANEOUS

(106) **Pioneers of interplanetary rockets.** E. BIALOBORSKI. *Przekroj* (268) (1950). (*In Polish.*)

(107) **Start into space.** *Natur u. Technik*, **4**, 295-298 (*Nov.*, 1950). (*In German.*)
The first International Astronautical Congress.

PHYSICS

(108) **Photographic study of surface-boiling heat transfer to water with forced convection.** F. C. GUNTHER. *Trans. Amer. Soc. Mech. Engrs.*, **73**, 115-123 (*Feb.*, 1951).
High-speed high-resolution photographic studies. An empirical equation for burnout heat transfer rate is derived.

PROJECTILES

(109) **Equations of motion of a rocket.** F. R. GANTMACHER and L. M. LEVIN. *N.A.C.A. Tech. Memo.*, 1255, 21 pp. (*April*, 1950). [Trans. from *Prikl. Mat. i Mek.*, **11**, 301-312 (1947).]
Derivation of the fundamental equations of motion of a rocket taking into account the external forces, the reactive forces and the Coriolis forces.

(110) **Missile guidance.** G. A. LONG. *Navigation*, **2**, 290-294 (*Dec.*, 1950).
After a discussion of the required frequency response of control the A4 guidance system is described.

(111) **Project reach.** C. DE VORE. *Navigation*, **2**, 275-281 (*Dec.*, 1950).
Description of the Viking rocket and the firing of No. 4 from the U.S.S. Norton Sound on May 11, 1950. Details of stabilization after cut-off by means of nitrogen jets, methods of data recovery by means of instrument salvage and by telemetry, and experimental instrumentation are given. The principal experiment conducted on Project Reach was the measurement of the directional, penetrating and ionising properties of the primary cosmic radiation at the geomagnetic equator.

(112) **Altitude record.** *Mech. Engng., N.Y.*, **73**, 138 (*Feb.*, 1951).
Viking's 107-mile flight.

(113) **Early rocket research.** *Mech. Engng., N.Y.*, **73**, 136-138 (*Feb.*, 1951).

(114) **The first large rockets.** W. LEY. *Weltraumfahrt*, **2**, 17-18 (*Feb.*, 1951). (*In German.*)
Brief description of some large German rockets of the late 17th and early 19th centuries.

ROCKET MOTORS

(115) **Thrust augmentation by air of exhaust streams.** E. SÄNGER. *Ing. Arch.*, **18**, 310-323 (1950). (*In German.*)
Considers the application to rockets, turbo-jets and pulse-jets. Concludes that the rocket exhaust velocity is too high for any useful augmentation to be obtained.

(116) **Rockets and assisted take-off.** A. V. CLEAVER. *J. Roy. Aero. Soc.*, **55**, 87-109 (Feb., 1951). [For extended digests see abs. Nos. 86 and 87.]

The basic method of operation of the bi-propellant rocket motor is described together with propellant performance and characteristics. The practical problems of rocket design with reference to propellant choice, general engineering problems and heat transfer are next discussed. Finally the application of rockets to assisted take-off is dealt with and details of the Sprite and its performance are given. Comparisons are made with other forms of take-off assistance and the advantages of rockets listed.

(117) **New technique for obtaining heat-transfer parameters of the wall and combustion gas in a rocket motor.** M. E. ELLION. *Trans. Amer. Soc. Mech. Engrs.*, **73**, 109-114 (*Feb.*, 1951).

This paper is principally concerned with the heat transfer through the rocket wall. A test method is discussed for obtaining gas side wall temperatures in a rocket motor in order to evaluate the suitability of various alloys for rocket-motor construction. Eight alloys have been studies by employing a thick-walled water-cooled nozzle into which cylindrical specimens could be inserted for tests. The heat flows and wall temperatures have been determined by employing the nozzle as a heat meter and by using a new calculation method that accounts for variable thermal properties with temperature. A modification of this technique is suggested for measuring the combustion-gas film coefficient in a rocket motor.

(118) **Measurement of thrust and movement on a small rocket.** H. GARTMANN. *Weltraumfahrt*, **2**, 11-16 (*Feb.*, 1951). (*In German.*) A 109-548 rocket motor was mounted so as to measure thrust and movement in all directions by means of electrical pickups. Maximum thrust was reached in 0·01 secs. after injection of the second propellant. The helical propellant tanks produced no turning moment.

ROCKET PROPELLANTS

(119) **Hydrazine compositions.** SOLOMON. *U.S.Pat. No.* 2,521,026.

Rocket fuel consisting of hydrazine mixed with H_2S or HCN. Advantages over hydrazine-water mixtures are lower melting points and higher specific impulses.

(120) **An approximate method for estimating the performance of oxygen-oil rockets.** A. B. P. BEETON. *A.R.C. Reports and Memoranda, No.* 2389, 15 pp (1950).

A method is described which enables the operating conditions of an oxygen-oil rocket to be determined to a reasonably accurate degree. Less important components such as OH, O and H are not allowed for, but the method is otherwise exact and could be extended to cover these components. Curves are given for the total heat and entropy of the constituent gases, and for the relevant equilibrium constants. These values are used to derive the total heat-entropy diagrams for four mixture ratios over a range from 1 to 60 atm. pressure in each case. Further curves are given which show the theoretical throat and outlet areas, and also the specific impulses, for expansion down to 1 atm. pressure.

(121) **Lithium as a suggested rocket fuel.** L. J. GRANT. *J. Space Flight*, **2**, (10), 3-5 (*Dec.*, 1950).

Discusses some of the problems of handling and using lithium.

(122) **Questions of space-flight.** F. Hoefft. *Natur u. Technik*, **5**, 13-16 (*Jan.*, 1951). (*In German.*)

Propellants of the future.

UPPER ATMOSPHERE

(123) **A suggestion regarding the use of rockets to vary the amount of atmospheric sodium.** D. R. BATES. *J. Geophys. Res.*, **55**, 347-349 (1950).

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REVIEW

A Handbook of Space Flight

(Compiled by Wayne Proell and Norman J. Bowman. 185 pp., price 28s., published (lithoprinted) by Perastadion Press, Chicago, U.S.A., 1950.)

From time to time—the most recent occasion was at the last Annual General Meeting—the suggestion is made that the B.I.S. should collect all the useful data relevant to astronautics and publish them in the form of a handbook suitable for ready-reference. In the work now under review, the two leading members of the Chicago Rocket Society have made a praiseworthy attempt to satisfy this same demand. In judging the results of their effort, it is only fair to keep in mind the magnitude of the task involved, though it is also true that anyone bold enough to undertake it naturally invites the criticism of specialists in a great number of fields. This review, in fact, is really a digest of comments received from various sources within the B.I.S., all of whom may be relied upon to have looked at the volume with the specially critical eye reserved by experts for works on their own subject!

The information given in the Handbook seems to be reliable in more cases than not, but this is not to say that it is entirely above reproach. For example, the quotation of a wave-length for cosmic rays (Table 18) is in disagreement with modern ideas on the subject, since the primaries are now known to be corpuscular, and the nuclear cross-sections given in Tables 47 and 48 are sometimes inconsistent. Also in the nuclear physics section, certain tables as printed are either meaningless, superfluous, or else display a definite confusion in the units employed. Again, certain of the astronomical data are given to either unjustified, or inconsistent, limits of accuracy. Turning to the tables concerned with rocketry, some of the data quoted for "Rockets of the World" is inaccurate in detail (e.g. the Vickers trans-sonic model did not use gasoline as its fuel); more seriously, the discussion of "maximum theoretical exhaust velocity," in terms merely of the heat of reaction of a given propellant combination, is of little more than academic interest. Some of the physical data listed for different materials (densities, melting points, etc.) seem to be doubtful, and in a few cases to have been transposed.

From the foregoing, it will have become evident that the book covers a very wide field, but it has to be studied at first-hand to appreciate just how needlessly wide! One is amazed, in fact, to find how much data on *rockets* the authors have omitted to include, while industriously collecting a mass of information on, for example, nuclear physics. Again, in a Handbook on space-flight, it is rather surprising to find values quoted for the mass of Eddington's universe and the radius of de Sitter's (two rival cosmologies, neither now generally accepted), and—at the other end of the scale—the conversion factor from the international to the absolute ohm!

In brief, we do not regard this work so much as *A Handbook of Space Flight* as a series of Tables of Physical and Astronomical Constants, about one-third of which might have some not-too-obscure connection with astronautics at the present time. Judged in this latter light, it could be a very useful collection,

if a certain amount of revision were done to correct erroneous and obsolete figures; we should also prefer to see references given to the original sources from which information was obtained.

From the "Dictionary of Words Used in Rocketry and Space-flight," given at the end of the volume, we learnt one new one, which appears to have some connection with the publisher's name: "perastadics" is defined as "the science and art of space-flight." This suggests a nicely alliterative motto for the B.I.S.—"*Per ardua ad perastadics.*" A. V. C.

RECENT AND FORTHCOMING MEETINGS

February 15, 1951 (Thursday). "Interplanetary Flight," by E. Burgess, to Manchester University Faculty of Technology Engineering Society, at the College of Technology, 4.30 p.m.

February 16, 1951 (Friday). "Into Space," by E. Burgess, to Davyhulme 18-plus Group, Chorlton House, Wilbraham Road, Manchester, 21, at 7.30 p.m.

February 19, 1951 (Monday). "Interplanetary Flight," by A. E. Roy, to the Hume Society of Edinburgh, at Fisher Institute, Warrender Park Road, Edinburgh.

March 29, 1951 (Thursday). "Into Space," by E. Burgess, to Chorlton 19-plus Group, Manchester, 21, at 7.30 p.m.

April 9, 1951 (Monday). "Interplanetary Flight," by E. Burgess. A public lecture organised by the Mayor of Hyde at the Town Hall, Hyde, Cheshire, at 8 p.m.

April 16, 1951 (Monday). "Rocket to the Moon," by C. Fleisher, to the Hendon and Golders Green Branch of the Association of Jewish Ex-Servicemen, at 8.30 p.m.

April 25, 1951 (Wednesday). "The First International Astronautical Congress." A short film and paper by E. Burgess to the Manchester Astronomical Society, Reynolds Hall, College of Technology, Manchester, 1, at 7 p.m.

May 1, 1951 (Tuesday). "Rocket Propulsion and its Aircraft Applications," by J. Humphries, to the Aviation Forum, Imperial Hotel, Victoria, S.W.1, at 7 p.m.

May 8, 1951 (Tuesday). "Rocket Propulsion and Interplanetary Flight," by A. V. Cleaver, to Yeovil, Som., R.Ae.S. Branch.

May 26, 1951 (Saturday). Extraordinary General Meeting of the Society at Caxton Hall to consider increased rates of subscription, at 6 p.m.

May 27, 1951 (Sunday). "Interplanetary Travel," by C. Fleisher, to the Mosaic Club, King's Hotel, Brighton, Sussex, at 7.45 p.m.

WANTED.—Copies of pre-war books on astronautics by Brugel, Noordung, Mandl and Ley. Also *The Moon* by Pickering. Send details of books offered and price to Box 10, BIS, 157, Friary Road, London, S.E.15.

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